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REVIEW OF RESEARCH RELATED TO LARGE WINDTUNNELS

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(or yaw) and roll, used routinely in their 3 ft, 4 ft, and 16 ft transonic, supersonic and supersonic windtunnels[14], a pair of high-load (4000 lb), high angle-of-attack (45°) forced oscillation mechanisms for roll and pitch (or yaw) is now under calibration or in an advanced stage of construction, respectively. Also at AEDC new test mechanisms have been recently

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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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A Further Review of Current Research Related to the Design and Operation of Large Windtunnels

NORTH ATLANTIC TREATY ORGANIZATION

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

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AGARD Advisory Report No.105

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A FURTHER REVIEW OF CURRENT RESEARCH RELATED TO THE DESIGN AND OPERATION OF LARGE WINDTUNNELS.

by

The Windtunnel Testing Techniques Subcommittee of the
Fluid Dynamics Panel

Third Report of a Series

This Advisory Report was prepared for the Fluid Dynamics Panel of AGARD

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SUMMARY

This is the third in a series of reports on research related to windtunnel design and operation. The first two were written by MiniLaWs (AGARD AR-68 and AR-83). This report is written by the Windtunnel Test Techniques (TES)* Subcommittee of the AGARD Fluid Dynamics Panel. Current results and planned effort for 346 studies and research investigations underway in nine countries are reviewed and commented upon in this report.

Part I describes the work of the TES Subcommittee and gives the rationale for the effort. Part II reviews the research that is underway and gives comments and recommendations derived from that review. These comments and recommendations are the principal contributions of the TES Subcommittee members. Part III summarizes the main conclusions and recommendations. Part IV lists titles, investigators' names and locations for the research and studies that are reviewed herein.

Four fields of work were given special treatment by the TES Subcommittee. In each of these four fields the TES Subcommittee appointed two conveners, one from each side of the Atlantic. Through the auspices of AGARD, these conveners brought together the foremost workers in each of the fields to discuss what needs to be done, how the work should proceed and how it should be shared. Seventy-nine leading research workers from nine countries participated in the work of the TES Subcommittee and made valuable contributions. Reports provided by the conveners are given in Appendices 4 through 7.

* the research committee shows in this report

The subcommittee is convinced that resources devoted to research related to windtunnel design and operation in the NATO nations are barely adequate,* ~~This report shows~~ that low redundancy and high effectiveness is exhibited by the program. As investigations discussed in this report come to fruition and the results are applied, there is an absolute need for the resources thus released to be used for acquisition of additional improved windtunnel technology in order to maximize the effectiveness of our limited resources. Needs for such advances and the possibilities for achieving the further technology gains are developed in this report and technology gains requiring further research are specified in Part III.

* TES – Technique d'Essais en souffleries, French equivalent of Windtunnel Testing Techniques.

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PART I

INTRODUCTION

This is the third in a series of reports on research related to windtunnels underway in the NATO nations. Brief discussions of the results currently being obtained from investigations on tunnel and model design concepts, test techniques and instrumentation, and fluid dynamics related to windtunnel testing are included. Comments are made regarding the adequacy of the efforts underway. Recommendations are made for further research methods of accomplishing needed work.

Reviews and evaluations of this nature to enhance the effectiveness of NATO technological efforts are considered to be a continuing requirement. Present demands for more precise and more types of wind tunnel data by advanced aircraft and weapon systems (AGARD-AR-60), major advances currently being made in instrumentation, test techniques, and tunnel design concepts (AGARD-CP174), and the necessity for windtunnel data for use in computational fluid dynamics (AGARD VKI Conference of June 1976, report pending) enhance the value of this particular review.

One of the means through which this report enhances the effectiveness of NATO wind tunnel research is by providing a means through which everybody working in the field is informed about what everybody else is doing. The reader may obtain additional information on any investigation mentioned in this report by contacting the TES* member in his country. Information exchange has also been promoted by TES through the 1976 AGARD Symposium on Wind-Tunnel Design and Testing Techniques and the 1977 AGARD Symposium on Laminar Turbulent Transition which were spawned by the work of the group. Add to this the cooperative study efforts, standardized model suggestions, and investigative techniques generated by the work of TES and its worth to NATO comes into focus.

In addition to the review and evaluation efforts, TES has brought about active collaboration between workers in four selected fields. In each of the four fields, TES selected two conveners, one from each side of the Atlantic. These conveners brought together the foremost workers to discuss what needs to be done and how the work should be done. The four fields and respective conveners are:

(1) Noise measurements in ground based facilities. R.Westley, NAE and J.Williams, RAE.

(2) Model design and its implications for the operation of pressurized windtunnels. S.A.Griffin, GD-Convair and J.Brocard, SESSIA.

(3) Design of transonic working sections. T.W.Binion, Jr., AEDC and J.P.Chevallier, ONERA.

(4) Transition in boundary layers. E.Reshotko, Case Institute and E.H.Hirschel, DFVLR.

The conveners' reports and recommendations are given in Appendices 4 through 7. Names of the experts that worked with the conveners are given in Appendix 3. Activities of the conveners have been extremely useful and productive and continued informal contacts between specialists initially brought together through this work is encouraged.

Titles of the investigations included in the program of work discussed in this report are given in Part IV together with their locations and the name of the associated principal investigator. Three hundred and forty-six investigations are reported herein compared to 308 in AR-83. Sixteen investigations were completed since the last report and there are fifty-four new inclusions. Current results, published reports, and planned effort for each investigation were obtained by inquiry of industry, universities, and establishments in nine countries made by the members of TES during the spring of 1976. Discussions of the research and comments on the recommendations from review of the research are given in Part II of this report.

Formulation of the discussion, comments, and recommendations under the several headings of Part II was shared out among the members. However, all content has been reviewed and reworked by all members so that the report represents the collective views of the committee. Dates and places of the meetings held by the subcommittee are given in Appendix 2. Main conclusions and recommendations are summarized in Part III.

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* Techniques d'Essais en souffleries – Windtunnel Testing Techniques Subcommittee. Names of the members of TES and their coworkers are given in Appendix 1.

PART II

COMMENTS ON CURRENT WORK

1. WINDTUNNEL DESIGN AND OPERATION

Design. As a consequence of current requirements for new facilities operating in the low-speed and transonic regimes, studies continue to be concerned predominantly with these speed ranges. Notable advances are being made in the construction programmes of large low-speed European tunnels. The RAE (Farnborough) 5 m tunnel (10.3)* and the ONERA F1 tunnel (10.7)[1]** have been pressure tested successfully; commissioning of both is scheduled to commence late in 1976 with model tests starting during 1977. Both tunnels use interchangeable model carts and can be depressurized locally at the working section.

Two proposed large atmospheric-pressure low-speed tunnels, the DFVLR GUK (10.9) and the NLR LST 8 m x 6 m (10.1), have now been merged into a single project known as the German-Dutch Windtunnel (DNW) (10.29). The tunnel will be sited at the North-East Polder where construction has now commenced; it will have interchangeable test sections 9.5 m x 9.5 m, 8 m x 6 m (closed and open jet) and 6 m x 6 m, and should be completed late in 1979.

After a 'cost versus capability' exercise ARC are now finalising the design of the modified 40 ft x 80 ft low speed tunnel (10.24).[2] The power of the drive system is to be raised to increase the maximum speed from 200 to 300 knots in the 40 ft x 80 ft test section and to furnish 110 knot maximum speed in a new 80 ft x 120 ft test section. A new fan with a low tip speed is being designed to pass on the extra power without degrading the noise level.

Experimental effort on transonic tunnels continues at ONERA (Toulouse) on the injector-driven tunnels T2 and T′2 (10.8),[3,4] at RAE (Bedford) on the pilot Evans clean tunnel (μECT) (10.2),[5] and at DFVLR on the Ludwieg tube;[6] the flow quality achievable with these drive systems is currently being assessed in the context of the requirements of the proposed European transonic tunnel. The plan to build a larger ECT at Bedford has been cancelled on cost grounds, and the studies in the μECT will be concluded for the present with the validation of some further improvements in the wave cancellation processes. Work on the pilot Ludwieg tube at AEDC has been terminated with the cancellation of the HIRT project.

Operation of the ONERA injector-driven tunnel T2 (10.8) was temporarily halted owing to structural problems but this facility was scheduled to be in commission again by the Autumn of 1976; a mass ratio of about 8 with a pressure ratio of 3.5 has been achieved at a test section Mach number of 0.9. A new test section with self-adjusting walls will be built in 1977. Injector performance studies continue at NASA Ames (10.14)[7] where mass ratios of 10 have been achieved at a test section Mach number of 1.0, with a pressure ratio of about 6. Further results are expected from tests using a square porous working section and a centre-line injector.

The transonic insert for the DFVLR Ludwieg tube (10.6) is complete and has been calibrated. The problems associated with the tube-wall boundary-layer growth have now been fully appraised[8,9] and DFVLR predicts that turbulent levels of 0.1% to 0.2% (well within the LaWs group specification) could be achieved by either using a diffuser, screens and settling chamber upstream of the test section, or increasing the diameter of the charge tube to lower the tube Mach number. The latter solution is to be preferred as the former is likely to shorten the run time by increasing the time for settling. Engineering studies on transonic tunnels to the LaWs specification, utilizing the three drive systems referred to above, are being evaluated under NATO auspices alongside a recently-completed study on a comparable fan-driven cryogenic tunnel.

LaRC continues to use their cryogenic transonic tunnel (10.15)[10] for validation experiments in support of the design of the US National Transonic Facility[11] which is based on the cryogenic concept and is scheduled to enter service in 1981. This facility will have a working section 2.5 m x 2.5 m and will operate at pressures up to 8.85 atmospheres. Theoretical and experimental studies of real-gas effects suggest that local liquefaction[12] can be avoided at the lowest operating temperatures envisaged, by close control of tunnel environment, and that any inaccuracies likely to be generated at cryogenic temperatures, by regarding the flow expansions and compressions as taking place in an ideal gas, would at worst be of the order of normal experimental uncertainty; further information relevant to increasing confidence on this important aspect is being exchanged between Europe and the USA in association with AGARD. FFA (10.23) in completing a conceptual study for a cryogenic blowdown tunnel using a high pressure storage and a heat exchanger have

* Parenthetical numbers refer to jobs listed in Part IV of this Report.

** Superscript numbers refer to references listed at the end of each section of this part (Part II) of the report.

shown that the project would be attractive from the viewpoint of the cost incurred to achieve a specific Reynolds number. ULICA (10.25)[13] have made a study of 93 wide angle diffusers in an attempt to formulate design rules. Further experimental work is proposed.

NAE (10.27) report failure by edge fatigue of the screens of the 5 ft x 5 ft blowdown tunnel. Redesign will call for greater reinforcement at the settling chamber wall/screen boundary. Control of Mach numbers in this tunnel (10.28) is the subject of an NAE design project with an intended Mach number accuracy of ±0.001 up to a Mach number of 1.4.

Methods of Constructing Rigid and Elastic Models. With (i) low speed pressurized tunnels entering service shortly at RAE and ONERA, (ii) the start of work on the US National Transonic Facility and (iii) the continuing studies on a European high Reynolds number transonic tunnel, emphasis continues to be given to model design and its implications for the operation of pressurized windtunnels. Useful progress is being made by a specialist Conveners Group on this subject, working under the auspices of the Sub-Committee sponsoring this report and its conclusions and recommendations are summarized in Appendix 5. The design study (12.1) for a model of a medium-range transport aircraft, suitable for the 5 m tunnel, was completed by HSA Hatfield in the Autumn of 1974. The detail design and construction followed on and this complex model with slats, flaps, control surfaces and 500 pressure measuring points is scheduled to be ready for testing in the Spring of 1977; this model is designed for use at total pressures up to 3 bars. The 3 m span calibration model designed and built jointly by RAE and ONERA, with blowing capacity and provision for engine simulators, is approaching completion and will be tested in the RAE 3 m tunnel and the F1 and SIMA tunnels at ONERA.

To the benefit of operators of atmospheric pressure tunnels, SSAB (12.3) have developed further and applied their technique for manufacturing large semi-span low-speed models by the numerically-controlled milling of thick aluminium plates glued directly to a honeycomb core. FFA (12.4) (12.6) have continued with their development of design, manufacture and testing techniques for scaled statically-aeroelastic models; results are to be published of transonic tests on a 1/30 scale model of an aircraft, showing good agreement with theory (panel method).

Flutter models, scaled in mass and stiffness, continue to be built at SAAB (12.7) IMF(L) (12.9) (12.10) (12.11) and ONERA (12.12) (12.13). DORNIER (12.5), in cooperation with IMF(L) and ONERA Modane, have built and tested a flutter model of the Alpha Jet.[14] The construction, particularly of the wing, is similar to that of the full size aircraft and combined techniques of chemical milling, fabrication and electron beam welding were used. Comparison of natural frequencies for model and aircraft showed that for the majority of modes studied, the model frequencies were at best correct and at worst within 6% of the aircraft value. The problem of making dynamically-scaled helicopter blades accurately continues to occupy RAE (12.8) who are now gaining some experience in the fatigue testing of model blades made in carbon fibre.

The static deformation of several models with differing kinds of construction has been studied at RAE (12.16). These represent combat aircraft configurations, have a range of sweepback angles, and were tested over a range of speeds and attitudes. The work is continuing with further models planned.

Review of Methods for Supporting Models. The high acceleration and deceleration rates of models encountering rapid incidence changes is likely to be of increasing concern with the trend towards larger model loads and shorter run times. NAE (13.3)[15] have established for a sting-mounted missile configurations in the 5 ft x 5 ft blowdown tunnel that, with certain precautions, satisfactory force and pressure measurements can be made at pitch rates as high as 15 degrees per sec.

Experimental techniques for the study of stores carriage and release are under development by BAC (13.8), who are recording the trajectory of stores released, using twin ejector guns, from a model parent aircraft; by AEDC (13.4) who are comparing, with free-flight measurements, results of windtunnel tests on external stores using internal and dual stings; by AEDC (13.6)[16] who have completed investigations of tunnel constraints on stores separation which suggest that there is general agreement between tunnel and calculated store trajectories.

ONERA is developing a six-degree-of-freedom support for captive trajectory studies of stores in the vicinity of the parent aircraft (S2MA windtunnel) and also a four degree of freedom vertical support with tilting head for studying the flight mechanics of conventional or V/STOL aircraft, with simulation of ground effects and gusts, in the SIMA windtunnel (13.2).

Sting-model interference has been studied by JPL (13.5) who have made sting and free-flight tests on cones in the ARC 6 ft x 6 ft tunnel; free-flight base-pressure measurements failed however to explain the increase of drag when the cone was sting mounted and this project has been terminated. NLR (13.7) have been concerned also with the analysis and quantification of support influence on model characteristics and are preparing a report on the subject.

In connection with the development of cryogenic facilities, a feasibility study of magnetic suspension for large scale testing was carried out at the University of Virginia.[17] Superconductor techniques and refined design could result in very low energy consumption e.g. less than 300 kW for a suspension system of appropriate size for the US National Test Facility. Installation of a magnetic suspension system is planned in the research cryogenic windtunnel at LaRC.

Methods for Data Acquisition and Analysis. Standard facilities for the processing of windtunnel data are being progressively extended and refined. A multi-channel system with quick-look facility (14.3) is now installed at FFA and is being tested in the S4 tunnel; a program library is being assembled for it. RAE (14.4) reports the gathering of useful performance figures from a prototype of the data system designed for the 5 m tunnel.[18] This prototype will continue to be used for the 13 ft x 9 ft tunnel and the complete multi-computer system for the 5 m tunnel, based on PDP 11/40 computers, is scheduled for completion by the time the tunnel is commissioned. ONERA (14.8) has improved the data acquisition and reduction systems of the Modane Test Center,[19] with on-line display of reduced data in the control room, and test processing control by local minicomputers connected to the central one.

Unsteady data analysis facilities continue to be operated successfully at RAE (14.5) and by NLR (14.9)[20] who are developing a new system for their own use. Improvement and updating of the NAE data acquisition facilities is planned by extending sub-routines (14.10) (14.11) to handle non-standard tests and by adopting a replacement data system based on a PDP 11/55 for use in the 5 ft x 5 ft blowdown tunnel. Tunnel operating time to document the trimmed performance of a model has been cut by a factor of 4 at AEDC (14.12) where a computer-controlled closed-loop trim system has been installed and operates in 5 degrees of freedom.

AEDC is operating telephone line connections between its data processing computer and the computer and information display equipment at three of its user installations. Data available in the control room can be communicated promptly to the users in Florida and Ohio, demonstrating the efficient utilisation of a single test center by a number of users.

Unconventional Design of and Alternatives to Windtunnels. There appears to be very little activity either in broadening the concept of the windtunnel or in adding to the known methods used to gain aerodynamic knowledge. However, work continues on a large low speed tunnel for gust studies, using catapulted models, IMF (15.3), and the sled at the Holloman track (10.18) has been used at sustained transonic speeds. The flow local to the model position in the latter has now been checked for flow angularity and the model pressure distributions have been compared with windtunnel data. The Aero-Train Bertin is being used for engine noise measurements with forward speed effects included (15.4).[21]

Hottner (Technical University, Stuttgart) (10.26) has proposed a hybrid technique to save tunnel drive power, utilising a model track. The model is propelled along this by a linear motor, thus reducing the flow velocity necessary in relation to a conventional wind tunnel.

Investigation of Techniques for Managing Turbulence in Windtunnels. There is little to report although NAE (16.5) (16.6), in their endeavours to discover the reasons for the screen failure in the blowdown tunnel (10.27), have explored the flow in the settling chamber just upstream of the screen positions. This revealed a level of turbulence dependent on the position of the air control valve. The pilot tunnel is being used to establish ways of improving the flow and reducing its destructive powers. NAE intends to carry out a flow check programme before and after the installation of the redesigned screens. Research on turbulence is covered further in Section 6 (Fluid Motion Problems).

Design of Transonic Working Sections. This subject is covered by a specialists Conveners Group. Their assessment of the current position is given, along with conclusions and recommendations, as Appendix 6. Special problems of testing at transonic speeds are covered more generally in Section 5.

Design of Anechoic Windtunnels. Concern over the external noise of aircraft and its alleviation continues to stimulate facility development work in several countries. The simulation of acoustic conditions at speeds up to 100 m/s with a 1.6 m diameter free jet will be possible in the new facility being built at CEPr (17.2); two larger jets will be available also with lower speed ranges. Experiments are being conducted at NLR (17.3)[22] on a 1/10 scale model of the LST (see 10.1 and 10.9) to determine criteria for the treatment of corner vanes and drive fans; more general aerodynamic and acoustic behaviour studies are also in progress.

RAE continues (17.1)[23] with their work aimed at increasing the usefulness of the 24 ft tunnel for acoustic testing. Two small facilities are being used for experiments. The first, a 0.43 m diameter tunnel has been used to look at the self noise of acoustic splitters, and the results of this work have been used to design the splitters for the second small facility; and 1/5 scale model of the 24 ft tunnel. In this model tunnel, predictions related to noise levels will be validated with a view to the improvement of the larger tunnel.

Aircraft-noise model-testing in ground facilities is covered by a specialists Conveners Group, whose conclusions and recommendations are summarised in Appendix 4.

Investigation of Real-Gas Effects in Air Flows at Sub-Ambient Temperature. Apart from the work carried out by LaRC using nitrogen as the flow medium (10.15) there are two more items listed of indirect relevance. ARC (1.12.1) are studying the feasibility of achieving high Reynolds numbers using heavy cases such as Freon 12 and argon whilst AFFDL have completed a study on the effects of water vapour on wind tunnel flow parameters.

Conclusions and Recommendations. It is perhaps inevitable that emphasis continues to be placed on problems requiring early solution. Design for new wind tunnels operating at high Reynolds numbers, the endeavours to ensure

that they have high flow quality, and the several projects to guarantee the integrity of the model, are all well established. There remains concern however about the feasibility of making flutter models for use at high stagnation pressures and, more generally, on the long fabrication lead time and high cost of the more complicated windtunnel models. The development and updating of instrumentation and data processing continues also in a healthy manner at all major test centers. However, there is reason for concern that effort is limited or non-existent in some areas, for example: acoustic resonance effects in large tubes; scaling laws for wave motion in non-uniform ducts. These can be recognized as fundamental problems of importance to the design of future generations of wind tunnel, conventional or otherwise.

The influence of the supporting sting on the flow about three-dimensional models has long been regarded with suspicion and the amount of work to quantify its influence is significant. This problem is of growing difficulty with the demand for high angles of attack in transonic testing for combat aircraft models. Endeavours to support models with the maximum of safety and the minimum of constraint are of major importance, and attention needs to be paid to the special requirements of those models having to contend with high dynamic pressures.

It is gratifying that there is much international co-operation in evidence, and it is to be hoped that this will continue to develop towards the increasingly efficient deployment of effort in this important area.

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2. GENERAL TESTING TECHNIQUES

Techniques for Measuring Steady and Unsteady Pressures and Forces.

(a) Pressure Measurements

Satisfactory instrumentation for steady pressure measurements is well developed with various degrees of sophistication in the transducer type and in the acquisition system at the different windtunnels.

The technical problem is more difficult for unsteady pressure measurements. Two methods have been used in the past:

- piped systems with a single or very few transducers (21.1), and
- multiple "in-situ" transducers (21.3).

Piped transducer systems are commonly used by NLR and DFVLR and the second method is established as a routine at RAE and ONERA (21.3).

Technological progress is reported in the development of subminiature transducers (21.9), mainly for use inside turbo-machinery.[1]

Attention is being given to the analysis of the flow inside air inlets with pressure measurements being used to obtain the mass flow, pressures and distortion (steady and unsteady), and the drag. Reference 2 gives a detailed presentation of various methods for making steady state measurements. An "on-line" view of the pressure measurements during the tests is needed to be able to modify the test program during a run. Equipment has been developed at ONERA[3] and at FFA (21.22) which provide "on-line", a pictorial view of the pressure distributions for use in modifying the test program during a run. A detailed map of the flow at the compressor face is also obtained by a movable automatic probe at KTH (21.20).

Unsteady engine inlet distortion is important, especially for highly maneuverable aircraft. A pressure map at the compressor face (frequency of about 1000 Hz at full scale) during about one second seems necessary to predict the engine behaviour behind a given inlet,[6,7] a representative map requires about 40 simultaneous pressure measurements, and such equipment is already operational in US Laboratories and at ONERA/Modane Center and is under development at FFA (21.6).

ARA (21.21) has developed a system for determining the cowl drag from wake pressure measurements. A movable pressure probe for use around inlet lips has been developed.

Work is underway on several applications of unsteady pressure measurements in rotating machines for applications to compressor blades (21.9), propellers (21.23) and helicopter rotors (29.1).

(b) Force Measurement

Significant progress has been made on calculation methods for heavy duty balance design using finite element computer programs. This is necessary for the increasingly severe environment encountered in new pressurized tunnels and/or at high angles of attack. This important problem is well covered in a special TES Conveners Group Report on Model Design[6] (Appendix 5), and many laboratories are concerned with this activity (see 21.14 for AFFDL).

Special rigs have been developed to analyze aircraft/weapon separation using the captive trajectory system (23.17) (23.18).[7] An AGARD FDP Working Group is studying store effects on aircraft performance.

Techniques for measuring and analyzing steady and unsteady flow fields. Flow field analysis can be made either as a survey of the complete flow field surrounding the model or a survey of the flow on the surface. Local measurements as well as visualizations are used and supplement each other in both cases.

Considerable progress in recent years has been made on various "applications of non-intrusive instrumentation in fluid flow research" as reported during the AGARD/FDP Symposium at St-Louis in May 1976 (Ref.8).

The application of laser interferometry to analyze boundary layers, free jets and noise in shock tubes is described by ISL (22.7). In particular a new method of double exposure streak-recording has been developed and used for studying unsteady supersonic jets. A new phase coupling technique has been introduced to laser interferometry which provides a local and absolute density and pressure record in the ultrasonic field near a free jet.

Laser anemometry has been rapidly developed during recent years and is widely applied in many aerodynamic flows, such as windtunnels, jets, flames, compressors. Velocimeters based on the interference fringe configuration are now into operation in several laboratories: AEDC, ISL, RAE, Imperial College, DFVLR, ONERA, etc.

At AEDC (22.24) two types of two simultaneous velocity component velocimeters have been built. One type is a forward scatter, two moving fringes, single color device. The second type is a coaxial backscatter, static fringe, two color device. Signals from the sensors are processed by a computer counting technique.

At ISL (22.9) various studies used forward scattered light velocimeters with acousto-optic modulators for velocity sign determination. Signals from the sensors are processed by a special counting technique. The LDV data are fed into computers which made possible correlations with measurements from other probes such as hot wires, microphones, pitot tubes, etc.

At RAE (F) (22.12) a two color velocimeter has been built to be used in general applications for windtunnels; the signals are processed by a photon correlator.

At DFVLR (P–W) (22.13) laser Doppler velocimeters were developed in a close cooperation with ISL for measurements in transonic and supersonic windtunnels. A transient recorder stores the basic data which is then read by an on-line digital computer to process the signals. Velocity data have been compared with the electronic counter system of ISL. A laser dual-focus velocimeter has been developed for compressor applications and windtunnel applications.

At ONERA (22.31) an operational velocimeter has been developed for use in a wide range of applications; including wind-tunnels (subsonic, transonic, supersonic), free jets, flames and compressors. Different components of the velocity are successively measured with their sign, through the use of acousto-optic modulators. Signal processing is accomplished by the counting technique developed by ISL.

At the Imperial College (22.20) various types of laser velocimeters have been tested. Integrated optical arrangements are used to investigate fringe, reference-beam, or single beam models of LDV with either forward or back-scattered light. Signal processing devices used by Imperial College include spectrum analysis, frequency tracking demodulation and counting.

At the University of Kent (22.11) the velocimeter measures high velocities and turbulence by a direct spectral analysis of Doppler shifted laser light using a static cofocal Fabry-Perot Interferometer. A single Fabry-Perot interferometer is also used at Kent in a two component laser anemometer.

Boundary-layers and shock-wave-boundary layer interactions have been investigated at AEDC, ISL, ONERA, RAE, DFVLR, and the University of Kent through use of their LDV systems. Measurements of turbulence in cold free jets and in hot free jets have been performed at AEDC, ISL, ONERA, and the University of Kent.[8]

Laser anemometry has been applied to high temperature flows and flames at ISL, Imperial College, and ONERA. ISL studies concern a combustion chamber. Studies at Imperial College apply the LDV to a plasma jet and to a 2 m. square furnace. ONERA has established a map of mean velocities and turbulence parameters in an air-methane flame at 2200°K using LDV.

Satisfactory results have been obtained using LDV in compressors at DFVLR (map of the velocities between the rotating blades) and at ONERA (detailed map of mean velocity and turbulence parameters of the wake downstream of a compressor disc).

Imperial College is using its LDV in the development of turbulence models.

At ISL laser Doppler velocity measurements in flight are planned in which the flow in the vicinity of the trailing edge of an airfoil will be studied.

A joint program of the US Army/NASA has been devoted to the study of helicopter rotor aerodynamics using an L.V. system employing the two color dual-beam backscatter operating principal in the 7 x 10 ft Ames tunnel. This L.V. system is capable of simultaneously sensing two components of the velocity around the rotor blade, and has given the radial distribution of the circulation around the blade and the tip vortex roll-up on the advancing blade.

A joint ONERA/ISL program in the S-3 Modane Tunnel has been devoted to making operational an LDV system which, after a final calibration study, will be used for transonic flow analysis around two dimensional airfoils (boundary-layers, shock-waves, wakes. . .)

Thermodynamic characteristics of freon have been determined at ONERA by measuring velocities in a Laval expansion nozzle, with a 0.5% precision using an LV.

Holographic interferometry has been extensively used to study shock-wave turbulent boundary-layer interactions in a transonic flow at ONERA (22.32); a great number of velocity profiles in the dissipative region have been measured from the processing of holographic interferograms giving precise density values.[10]

Laboratory work is under way at AEDC (22.27) on a video digitizing system for holographic interferograms, and it appears that reflected diffuse light interferometry provides a unique technique to observe flow fields in a cavity.

Two types of Raman effect instruments are under active development for aerodynamic flows and flames.[11] One type is Conventional Raman scattering which is simpler to implement (AEDC 22.26) but of limited sensitivity if stray light is present. The second type is Coherent Raman scattering which is more delicate and expensive. Research efforts on this type are in progress in US (USAF/Aeropropulsion Lab, WPAFB; Naval Research Lab., Washington DC; Sandia Lab.) and at ONERA (22.33).

Well known techniques for flow visualization at low speed by smoke or bubble are now being tentatively applied at high speed. The smoke technique is mainly used at DFVLR (G) (22.5) to make visible the lee side vortices of bodies of revolution for $M = 0.5$ to 2.2. The helium-filled bubble technique is used at Sage (22.15) in a transonic tunnel to obtain both streamlines and recorded bubble velocity information around models.

USAF laboratories have developed infrared techniques for measuring model surface temperatures. At AEDC and NAE (22.5) (22.34) an infrared scanning camera is being used to determine boundary layer transition, heating data and thermophysical properties of materials for aerodynamic heating models in transonic and supersonic tunnels.

At AFFDL (22.28), an infrared pyrometer for model surface temperatures is used in their pebble bed heated tunnels. Data are taken during model heating.

Measurement of surface shear (skin friction) is obtained by Oxford U. (22.14) with a floating element method in a study of roughness effects. The Preston-tube method is used by FFA (22.23) for measuring skin friction with miniaturized probes. KTH (22.30) has also developed a flow direction probe used for low speed tunnel tests.

Hot-film transducer technology, developed originally by USAVLAB (McCroskey) is now operational in several laboratories. At the USA ARL-Ames and at ONERA (22.8; 22.22) this technique is used on oscillating airfoils and on helicopter blades for studying in real time the flow history (laminar turbulent, separated or reattached, dynamic stall, etc.), including shock-boundary layer interaction at transonic speeds[12]. This latter application is also utilized at NLR (22.29).

Dynamic Stability Testing. As a result of the greatly increased interest in dynamic stability problems in recent years[13], several advanced windtunnel techniques and new experimental arrangements are being developed by various organizations in both Europe and North America. Among problems requiring immediate attention, probably the most important is the occurrence of large nonlinear variations with angle of attack in most of the dynamic stability parameters. This effect, which occurs at angles of attack high enough to cause flow separation and asymmetric – often unsteady – shedding of vortices from long pointed bodies, is known to cause abrupt changes, sometimes of an order of magnitude and often involving a change in sign, in many derivatives. This includes the important primary damping derivatives in pitch, yaw and roll, and applies to both aircraft and missiles. Except for the forced-oscillation pitch, yaw and roll apparatus in the LaRC Full Scale Windtunnel ($M < 0.1$), no experimental arrangements existed until recently for measurement of dynamic stability derivatives at higher angles of attack. This situation is now being remedied at several laboratories. At AEDC, in addition to an earlier developed pair of dynamic balances for forced-oscillation in pitch

(or yaw) and roll, used routinely in their 3 ft, 4 ft, and 16 ft transonic, supersonic and supersonic windtunnels[14], a pair of high-load (4000 lb), high angle-of-attack (45°) forced oscillation mechanisms for roll and pitch (or yaw) is now under calibration or in an advanced stage of construction, respectively. Also at AEDC new test mechanisms have been recently developed[15] for obtaining dynamic stability parameters in pitch and roll on missile models at angles of attack up to 90°, using free-oscillation and free-rolling techniques. NAE (23.20), in cooperation with ARC, is developing a series of forced-oscillation balances for studying oscillations in various degrees of freedom at high angles of attack. One such apparatus, for oscillation in pitch or yaw, has already been used in windtunnels at both NAE and ARC for experiments at angles of attack up to 40°, and at angles of sideslip up to 10°. A continuous-rolling apparatus is being designed at BAC (23.21) for use in both low-speed and high-speed windtunnels in the UK. Several free-and forced-oscillation balances are being routinely used at ONERA (23.11; 23.13).

The same flow phenomena which are responsible for the highly nonlinear effects in the damping derivatives at high angles of attack, are also responsible for significant aerodynamic coupling effects between the various degrees of freedom. In addition to the traditional cross derivatives pertaining to yawing and rolling, a new category of cross-coupling derivatives has now emerged, relating the longitudinal and lateral degrees of freedom. As correctly realized in the past, these cross-coupling effects do not exist at low angles of attack, when the flow remains symmetric; however, they can no longer be neglected in the presence of flow asymmetries at high angles of attack or in the presence of sideslip. Significant cross-coupling derivatives such as yawing and rolling moment derivatives due to pitching or pitching moment derivatives due to yawing have now been measured with the cross-derivatives apparatus developed at NAE (23.20).[16] A special three-dimensional calibrator for the apparatus has also been developed.[17] Cross-derivatives can be obtained with the AEDC and ONERA equipment mentioned before. Cross-derivative balances also exist and are being continously developed at RAE (23.15) and DFVLR (23.3). At this latter organization several new dynamic balances for use in both low-speed and transonic windtunnels are being designed and constructed.

In connection with the new concepts of direct-lift and direct-side-force controls, there is an increasing interest in dynamic derivatives due to vertical and lateral acceleration. A half-model balance for vertical acceleration derivatives[16] and a full-model apparatus for measuring moment derivatives due to both vertical and lateral acceleration are being developed at NAE. This type of information is also required to separate the purely rotary derivatives from their oscillatory counterparts; more work along these lines is being carried at LaRC[18] and at VPI, where purely rotary derivatives are measured at low speeds in a curved-flow test section.[19]

In recent years there has been an increased emphasis on a better simulation of the aerodynamic phenomena that are associated with the spin motion of aircraft. Also, it was shown[20] that to take into account the non-linear coupling effects that exist between pitch, yaw and roll, a generalized formulation of equations of motion was necessary, and that in this new formulation one of the important contributions to the total aerodynamic moment was related to the rotary or coning motion. To simulate such a motion in a windtunnel the model, at some fixed combination of incidence and sideslip, is attached to a rotary balance, whose axis is parallel with the windtunnel centerline. Several such balances have recently been constructed or are being designed, for both low-speed and high subsonic windtunnels, including those at LaRC, ARC, RAE(B) (23.7), DFVLR (23.3), and IMF (23.14). A good discussion of ARC's activities involving the use of a rotary balance can be found in Reference 21, where a description is also given of the new large-scale ARC rotary apparatus for use in the 11 ft and 12 ft wind tunnels. This new apparatus, which is now being assembled, will allow a remote change of angles of attack and sideslip, up to a combined value of 30°; the use of bent stings and top-mounted models will permit a further adjustment of the angle of attack to 100° and of sideslip up to 25°.

In addition to the usual aerodynamic static interference, a sting used in oscillatory experiments introduces a dynamic interference due to its oscillation. This affects both the magnitude of the data measured and the true position of the center of oscillation. A recent assessment of this problem has been made at AEDC.[22] One way, of course, to avoid any sting interference is not to have any sting at all. At low angles of attack this can be done by performing dynamic testing employing the half-model technique or, alternatively, using magnetic suspension. Recent applications of the half-model technique have been reported by NAE,[16] where measurement of vertical acceleration effects is being prepared, and by AEDC (23.19), where experiments in the 1 ft transonic tunnel using a new dynamic half-model balance are being planned. The half-model technique is also eminently suitable for experiments involving two simultaneously oscillating models[16] or experiments including simulation of the jet exhaust plume behind an oscillating model.[23] The application of the concept of magnetic suspension to dynamic testing has been pursued by several laboratories, including the University of Virginia.[24]

When performing dynamic stability testing, special consideration has sometimes to be given to conditions during take-off and landing. Dynamic experiments in the presence of simulated ground effects are being planned at ONERA (M) in the S1 MA Wind Tunnel. (27.8).

In addition to dynamic stability characteristics of a rigid aircraft, dynamic derivatives due to oscillating control surfaces are also of interest. Such measurements have been performed with a special balance at RAE (23.1).[25,26] In this connection the unsteady pressure distributions due to an oscillating control surface have also been determined, using Kulite miniature pressure transducers.[27] Similar experiments are also being performed at ONERA (M).

A comprehensive review of the North American equipment for dynamic stability testing has been published by ARC.[28] An AGARDograph on the same subject is in preparation.[29]

The measurement of aerodynamic and structural damping, and of the frequency response to disturbance. One of the main goals is the identification in a wind tunnel (or in flight) of the unsteady aerodynamic components.

An interesting new method developed at ONERA[30] consists of obtaining, on-line, the transfer function of the model for many modes with the excitation source being the deflection of the rudder (with a servo-Jack). Excitation can be of the form of a sweep frequency or a white noise. A Hewlett Packard 2100 computer is used to obtain these transfer functions from which the complex roots and residues are obtained.

DFVLR(G) (24.2) is using a hybrid computer and several types of excitations (harmonic, stochastic, frequency sweep, and transient . . . .) in tests which give eigen frequencies and damping characteristics. NLR/ONERA (24.1) and RAE using in real-time, a Fourier analyser to obtain frequencies and damping from the response of the model to the natural turbulence of the windtunnel.

Techniques for measuring aeroelastic and flutter characteristics. Aeroelastic (static and dynamic) deformations of a model during windtunnel testing can be a problem in the new pressurized tunnels for high Reynolds numbers. Care will be required to take advantage of such deformations in securing aerodynamic data and in avoiding unsafe structural conditions for model and support system. An accurate and reliable method for measuring the static and dynamic deflection of a model during testing will be mandatory. Three approaches are reported:

(1) At AEDC (25.8), encouraging results have been obtained for measuring remotely steady-state and vibratory deformations of small amplitude on selected discrete points of model or support system using two-beam laser interferometer concept with retro-reflectors on model surfaces.

(2) At Volvo (25.3), a method has been developed to determine natural frequencies and mode shape for dynamically correct model (SAAB-Scania) and a holographic device has been developed to study vibration mode shapes on structural dynamic flutter from time-average holographs.

(3) At ONERA (25.9) an optical technique is being developed for vibration measurements on turbomachinery blades using non-coherent light and a laser interferometer has been developed to measure the local vibratory deflection in each point of the blade.

Controlled excitation of the model for flutter testing is sometimes limited in frequency or frequency ranges, by the rig characteristics: excitation by tunnel turbulence is extensively used by NLR (25.1) and ONERA/RAE (25.6; 25.4) where several joint research programs were undertaken in the past few years. Three types of model suspension systems are in use for flutter testing:

(1) At NASA-Langley, a cable system is extensively used in the transonic dynamic tunnel.[33]

(2) At Boeing, a "yoyo" suspension system (universal joint) is used to support the model for low-speed testing. This method is also operational at DFVLR(B).[34]

(3) At ONERA(M) (25.6), the model is suspended, with five degrees of freedom around a fixed mass in space. This mounting system seems to minimize model suspension interference, mainly at transonic speed.

Techniques for simulating and measuring transient motions, such as gusts. The IMF(L) has under way studies of the response of a free-flying model (launched by a catapult) to a discrete vertical or lateral gust generated by the flow of an auxiliary open-jet windtunnel installed perpendicular to the aircraft model trajectory (26.1); load, acceleration, and pressure data telemetered from the model, and model motions picked up by TV cameras, are analyzed in real time during the free flight and recorded on magnetic tapes. A comprehensive description of the test rigs and typical results were given at the AGARD/FMP Meeting of Valloire in June 1975 (Ref.32). The technique has been used to study the validity of various analytical predictions of the wing/horizontal tail and wing/fuselage interaction with a given vertical gust shape. It is also possible to simulate a gusty approach in ground effect including lateral gusts (27.5 and 27.6).

Effects of gusts are also studied using a conventional windtunnel with special provisions to generate various oscillatory flow motions around a fixed aircraft model. This technique was first applied in the NASA/Langley transonic dynamic tunnel, using four oscillating lifting surfaces in the front of the test section.[33] At the DFVLR(B), a 2 dim. gust generator was developed in the front of the open test-section of their 2.8 x 3.6 m^2 low-speed tunnel (26.6) and used to excite an aeroelastic semi-free model with vertical gusts.[34] This gust generator consists of two wings with movable flap driven by an electrically controlled mechanical crank, generating either sinusoidal gusts up to a frequency of 10 Hz or various types of discrete or stochastic gusts. This rig will also be used for developing active control systems for gust alleviation on aircraft models. A similar technique, with a linked array of aerofoils across the upstream of the open jet of a small tunnel has been developed at the University of Salford, UK.[35] The random or sinusoidal frequency range of the Salford rig is up to 20 Hz and sharp-edged gusts can also be produced.

ONERA(Ch) has developed a technique for generating gusts in a pilot-tunnel using two oscillating jet-flaps in the front of the model (26.4). ONERA has plans to apply this technique in the large S1 Modane windtunnel as a part of the new flight mechanics rig under development. A similar approach has been used in a low-speed tunnel at the MIT, under NASA contract[36], with two wing sections having rotating nozzles at the trailing-edge to generate oscillating flow in the front of the model (axial or lateral sinusoidal gusts). This rig has been used to study the response of a helicopter rotor

model to various types of gusts. The USAF Aero Propulsion Laboratory is using a related approach employing unsteady fluidically controlled flapping jets for production of variable frequency gusts in a wind tunnel. This system is intended for tests of aircraft models or of turbomachinery components.[38]

RAE(B) (26.5) is still very active on a theoretical model study of various actual gusts encountered in flight and aircraft response.[37] Results indicate that, for the investigation of longitudinal handling characteristics, both isolated ramp gusts and sequential pairs of such gusts of opposite signs seem realistic, but difficulties are being encountered in achieving a satisfactory simulation of the aircraft response to turbulence of high intensity.

Techniques to measure ground effects. Correct simulation of ground effect in a windtunnel is still a difficult problem, even with new techniques developed to avoid a parasitic boundary layer on the test section wall or on the ground board. To avoid this problem, the "moving belt" concept was successfully applied in various facilities (RAE, LaRC, Vertol), but this technique is very expensive to install in a conventional tunnel and the speed is limited. On the other hand, boundary-layer control (either with suction or a blowing system) applied on the fixed ground plane seems an effective solution in many instances. A ground plane with uniform boundary-layer suction is operational at the DFVLR (G) low speed tunnel and was used to study ground effect on a VTOL model with deflected fan-flow (27.11).[6] A ground plane with two discrete blowing slots has been built for the ONERA-S1 Modane Tunnel (27.8), and will be operational at the end of 1976. Provision will be made for simulation of the dynamic ground flare in approach (vertical motion of the model towards the ground board). This blown ground plane will also be used for large half-model testing without boundary-layer separation on the reflection plane.

In the IMF-Lille rig with free-flight models launched by a catapult,[32] the flare is simulated with an adjustable ground plate (27.4). A special open throat tunnel built along the model trajectory is used to simulate lateral wind (27.5) and lateral gust (27.6) during approach. This rig is fully operational.

The ground effect is measured through pressure distribution measurements on a ground plane in the DFVLR (P–W) Tunnel and research is reported of the analysis of the flow field around cross blown lifting jets with various dynamic pressure ratios (27.9).

In the VKI Tunnel in two-dimensional flow, a wing section with flap has been tested at various ground plane altitudes to compare the ground effect measured and calculated. Boundary layer separation occurred on the ground plane without boundary-layer control Separation was subsequently prevented by suction applied through a perforated ground plane (27.10).

Methods for determining spinning characteristics. An increasing number of aircraft designs are now capable of sustained flight at very high angles of attack, where some degradation of flying qualities appears, followed by a fully developed spin. The subject is of such interest that it was recently covered in detail during an AGARD/FMP specialists meeting (VKI, November 1975; published as AGARD-CP-199, "Stall-Spin Problems of Military Aircraft").

Analytical spin prediction methods are being developed at the VKI (28.5) and in France.[41] However, a better mathematical approach for these highly non-linear regimes is still needed. Measurements of static and dynamic aerodynamic coefficients during spin can be obtained by use of a rotary balance (28.2). Despite the increased use of dropped model tests, the vertical tunnel is still a basic tool for spin studies extensively used at NASA (L)[40] and at the IMF (L), (28.1), (28.3).[32]

Large scale remotely piloted or preprogrammed models (RPRV) dropped from helicopters or aircraft are used at the RAE and at the US Edwards Flight Test Center[42] for spin investigations. Use of sophisticated instrumentation and telemetry, coupled with modern parameter identification techniques give a good aerodynamic description of the high angle of attack characteristics and of the spin development with a much more realistic Reynolds number than in present day spin tunnels. This method is very expensive because of the high cost of model fabrication and equipment, recovery procedures with parachute + helicopter, etc.

The design of rigs for testing rotary wings. Special rigs for rotor testing are in operation in numerous laboratories. Provisions are available for measurements, as well as sophisticated local analysis of pressures, unsteady loads (vibrations, stall, flutter, etc.) and for flow visualization and noise measurements.

The largest installation for full-scale helicopter testing has been in operation for several years in the 40′ x 80′ wind tunnel at NASA-Ames.[43] Three other US Laboratories are well equipped for complete model or rotor testing. There are the V/STOL tunnel at NASA-Langley, the Boeing-Vertol Tunnel[44] and the United Technology/Sikorsky Tunnel. In Europe the 24 ft RAE low speed windtunnel[45] is now used for rotors with dynamically scaled blades (29.3). The ONERA S1 Mondane (8 m) Tunnel is extensively used for large scale rotor models (helicopter or convertible) up to high-speed (29.2) with on-line acquisition of unsteady and steady loads, and stroboscopic flow visualization by threads or smoke (29.4). Transition from hovering to cruise flight, and vice-versa of a large tilt-rotor (5 m) has been simulated in real time in this tunnel with a fast variation of the tunnel speed and correlated control by computer of the corresponding rotor parameters (rotor inclination, general and cyclic pitch). Selected data were reduced and displayed in real time to the control room. The onera S2 Chalais (3 m) tunnel is used for research on small rotor models with force and pressure measurements on the blades, with realistic tip Mach numbers (29.1).

Numerous 2-dimensional rigs for simulating unsteady flow around a typical rotor blade section have been developed. Such rigs simulate special motions including pitch, plunging, and heaving for use in basic and applied research. Aerodynamic rigs of the type are located at AMRDL 7′ x 10′ Tunnel,[47] Boeing-Vertol,[49] UT-Sikorsky 2 dim. tunnel, ARA 2 dim. rig for development, ONERA S3 Modane, CEATS-S. 10 Toulouse and IMF Marseille. In addition a water tunnel is used by the US Army for steady and unsteady tests with flow visualization.[48]

Methods for measuring noise and development of noise generators. The problem of noise measurements in ground-based facilities with forward speed simulation is still a subject for active cooperative programs and work of a TES conveners' report given in Appendix 4.

The requirement for new large anechoic tunnels (discussed in Section 1 above) has led to the initiation of research to define the usable domain of noise testing and the necessary corrections to apply to acoustic measurements made inside and outside the test section of existing large wind tunnels. Several recent specialists' meeting have reported on these subjects.[50,55] Considerable work on noise measurements in the 24 foot RAE wind tunnel is still in progress (2 10.1).

At the VKI (2 10.3) a joint program with ONERA is aimed at a better interpretation of noise measurements made in and outside of the open working section of a windtunnel. The data are influenced by the passage of the acoustic wave through the mixing region of the open tunnel jet (convection, refraction, and diffusion effects). The ultimate goal of this research is to develop correction methods for the future measurements outside the free jet of the new anechoic tunnel CEPRA-19 developed in France at the CEPr-Saclay. Noise tests have been made in closed working sections of large tunnels: at S1 Modane, mainly around helicopter rotors (2 10.6) and in the 40 x 80 foot tunnel at NASA-Ames (2 10.8) around various full-scale aircraft. Techniques were developed to discriminate between the noise generated by the model and the extraneous noise generated by the tunnel (eight-element microphone array and two-element correlation microphone, to eliminate reverberant noise and microphone wind noise).

ONERA has undertaken basic research on the space-time structure of acoustic fields to study the narrow field of a free jet (2 10.9). RAE(F) has developed a modified Hartman-type air-jet noise generator (2 11.1) used for investigating noise shielding and flow field refraction effect in windtunnel experiments. NLR (2 11.2) has demonstrated the satisfactory acoustic simulation of a turbo-fan engine with a small model working with decomposed hydrogen peroxide.

A group of specialists assembled under the auspieces of this subcommittee has studied test section requirements and circuit designs for acoustic wind tunnels to provide anechoic testing environments. This group considered special measurement and analyses techniques for noise-model research and simulation of propulsion noise sources at model-scale. Results of their study are summarized in Appendix 4.

Techniques for simulating adverse weather conditions, such as icing, rain erosion, etc. Generally, techniques for simulating adverse weather conditions can be incorporated into conventional large wind tunnels for use in studying the behaviour of actual parts, or scaled-down models, of aircraft, rotorcraft or missiles. Correct simulation of phenomena such as icing, rain erosion, or decreased visibility due to rain, requires that similarity rules be satisfied, taking into account various parameters such as speed, temperature, run duration, droplet diameter, liquid water content, etc.[56]

Icing testing at large scale is common practice in the ONERA S1 Modane Tunnel (2 12.1), taking advantage of low atmospheric temperature in winter. Icing conditions are produced by means of a spray-grid ahead of the models. Good correlations with flight testing have been demonstrated including results on the Concorde slender-wing and nacelles. A comparison was recently made between ice accretions as predicted by a computer program (SNIAS) and those actually observed on a corresponding tail plane element tested in the tunnel. A quite good correlation was found within certain limitations. Icing tests were also performed on a large helicopter rotor.[57] In the same S1 Modane Tunnel, an artificial rain generator system is also available for investigating at full scale, the visibility through wind shields up to a velocity of 150 m/sec (Ref.58).

Rain effects on actual aircraft or missile components (leading-edge, radomes, etc.) are commonly performed in the transonic blow-down S3 Modane up to transonic speeds (2 12.2).

Mention must be made of the McKinley climatic laboratory facilities developed at the US Air Force Armament Development and Test Center (Eglin AF Base), where a huge insulated hangar (252 x 201 x 70 ft) can accommodate full-scale aircraft (like the C-5A). Extreme weather conditions including temperature and humidity, snow, rain, wind, etc. are simulated for long time periods for weapon systems certification.

Conclusions and recommendations. It is important to increase the cooperation between various organizations on the development of balances and support systems and on the correction of aeroelastic effects for the increasingly severe environment encountered in new pressurized tunnels and/or at very high angle of attack at transonic speeds. The same conclusion is valid for the dynamic stability problems which become increasingly important for any new highly manoeuvrable aircraft.

Increasing interest on the development of C.C.V. techniques, requires that flutter characteristics and gust responses be extensively studied at the preliminary stage of a project and new that testing techniques and sophisticated instrumentation must be developed.

A very significant effort was made during the last few years in various countries to develop new specialized facilities for noise studies with forward speed effect. Continuing cooperation is needed on the best use of such facilities, including basic acoustic research, measurements and analysis techniques and correlations with flight measurements.

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3. SPECIAL TECHNIQUES FOR ENGINE SIMULATION

Engine-Airframe Windtunnel Testing Methods. Tests with flow simulation for underwing nacelles have ben conducted at ARA (30.1). A report on the experience with free flow nacelles, blown nacelles, and powered nacelles, has been published.[1] In connection with tests on a 1/25 scale model of a typical wide bodied jet, a system of high pressure air ducting past a standard internal balance has been designed, using a combination of precision bellows and tubing which will duct two primary and two secondary airstreams.

Nozzle-afterbody tests are continuing at AEDC (30.2). The present situation is that, after continuing tests to determine the mounting strut influence on afterbody drag for strut mounted models, an analytical study has been initiated to determine the feasibility of simulating the jet plume using a sting mounted afterbody model with an annular jet.

An ejector driven engine simulator for tests in an aircraft model is being designed and fabricated. An experimental program to measure afterbody geometry effects on fore body pressure drag of an equivalent body of revolution was conducted in the AEDC 16T.[2,3] DFVLR (32.2) investigates the influence of hot jets on afterbody configurations: the following tests have been performed or being planned: Influence of jet parameter on boattail pressure distribution of an HFB 320 engine nacelle model, the influence of surface temperature on pressure distribution, influence of test section characteristics (open-closed) on pressure distribution of an AGARD model, and flight tests[4] (see also Section 6 (6 11.5 and 6 11.11).

A rig to measure thrust and afterbody drag has been designed by BAC (6 11.11). Single and twin nozzle afterbody configurations have been tested at supersonic Mach numbers over a wide range of jet pressure ratios.[5]

In the field of interference simulation scaling a new activity is announced by BAC (30.7). A jet lift model is investigated over the transition flight regime with the aim to find the validity of momentum scaling.[6] Further tests are planned in the Warton 5.5 m tunnel and in the RAE Bedford 13′ x 9′ tunnel.

The work on engine inlet testing at high maneuver conditions at AEDC (30.3) has been completed. The objective of these investigations was to improve the test capability to test full-scale inlet/engine configurations with forebody effects at transonic velocities. Tests have been carried out in a 1 ft tunnel to obtain design information for flow shaping devices to be installed in the AEDC 16T.[7]

Nozzle-afterbody thrust measurements are also in progress at ONERA (33.5). The rig is set up in the high subsonic windtunnel S3 of Chalais-Meudon (ϕ = 1 m).

The afterbody is fixed to an upstream sting and force measurement is provided. The boundary layer on the sting is reduced by using a blowing slot. Comparisons of ONERA and AEDC test results on an AGARD afterbody model show that reducing the sting boundary layer is pretty comparable to doing a test at the higher Reynolds number which would give the same boundary layer.[8]

A new test rig has been specially designed and is being built to study the afterbodies of the high by-pass ratio engines.

Work on engine/airframe exhaust system interaction at MDC (30.4) has been completed. The empirical program based upon F 4 J flight and windtunnel test data has developed techniques to configure each of the propulsion system elements (inlet, engine, and exhaust) for best total system performance.

Engine Simulators. A joint RAE-ONERA (31.1) note on the calibration of two ejector driven turbofan simulators for use in subsonic pressurized windtunnels has been published.[9] AFAPL (30.6) has constructed a multi-mission turbine engine simulator. It has been refined during development tests in the AEDC engine test facility. The engine simulator was tested in a nacelle in the AEDC 16T. Data are currently analyzed.[10,11] The simulation of rocket engine jets in small windtunnel models has been demonstrated by DFVLR (32.3). Solid propellants with a combustion chamber of 20 mm DIA can simulate rocket engine giving a run time of ~ 2 sec. By changing the design, a test time of ~ 3 set has been attained. In parallel a 300 bar pressure tank of 10 m^3 volume for secondary jets of very high total pressure and high mass flows has been installed.

Tests with an ejector driven RB 211 simulator (BAC (34.2)) have been carried out in the 13 x 9 ft low speed wind tunnel at incidence.[12] Design work on an RB 211 simulator for high subsonic speed applications is currently in progress. The tests of single and multiple nozzle ejectors as basis for design of ejector-driven engine simulators have been completed.

Two progress reports have been issued[13,14] on work on a simultaneous simulation of engine intake and exit flow at BAC (34.4). After having designed an ejector powered combat aircraft model tests on blade sting-afterbody interference are being prepared for the ARA wind tunnel.

DFVLR (39.1) has designed a small propane oxygen gas generator to simulate hot and fuel-rich primary jets. This gas generator located within the center-body of a ramjet allows many experiments which extend the burning range of the ramjet combustion chamber in the direction of lean air-fuel mixtures.

Instrumentation and balances. DFVLR (32.1) is trying to speed up data acquisition and reduction by a new on-line data reduction system (14.1). The aim is to improve parameter adjustment (e.g., mass flow through simulator) during test run. The plan of VOLVO (33.2) to investigate the use of pneumatic balance techniques of zero-lift drag measurements with simulated engine jets has been carried out. Preliminary tests using AGARD afterbody models mounted on a central body supported by a single strut have been run. The comparison between pneumatic type balance and strain gage balance

has shown superiority of the pneumatic balance due to the absence of temperature drift and smaller test point scatter. A double cylindrical shell balance has been developed by GD which permits simultaneous supply of a jet flow while allowing precise balance measurements.[15]

Conclusions and recommendations. Very extensive activities are reported on nozzle-afterbody-tests in the transonic flow regime. These activities are partly due to the Ferri proposal (AGARD afterbody model) but follow now their independent path. The large influence of aircraft performance of afterbody design has been demonstrated by many authors.[16] Afterbody drag measurements and jet plume simulation are the activities reported most extensively. Flight tests with afterbody models have been announced.

- The AEDC activity on engine inlet testing at high maneuver conditions should form the basis for engine simulation in complete models at completely separated (post stall) flow conditions.
- The comparison of flight and windtunnel tests of propulsion components (MDC) gives an important insight to the influences of the different components (inlet, engine, exhaust).
- The joint RAE-ONERA program on developing an ejector driven engine simulator was stimulated by the existence of two new pressurized low speed tunnels (ONERA F 1, RAE 5 m tunnel) has been completed successfully.
- One of the most advanced projects of engine simulation techniques at AFAPL has been finished (Multi-mission-turbine-engine-simulator). It can be expected that this simulator will be suitable for many applications.
- BAC continues to concentrate on ejector simulators. The design of an RB 211 simulator for high subsonic application is in progress and tests on single and multi nozzle ejectors have been completed.
- The application of a pneumatic balance technique as demonstrated by VOLVO seems to be successful due to zero temperature drift and low test point scatter.
- Effects of afterbody flow on forebody flow are bieng investigated but the source of discrepancies needs to be clarified. Effects of forebody flow on afterbody flow needs to be investigated.
- A method for the measurement of momentum and mass flux in engine simulators needs to be worked out.
- Possible flow field calculations should be applied to inlet flow as well as to exhaust jet flow.

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4. SPECIAL TECHNIQUES FOR HIGH-LIFT AND V/STOL TESTING AT LOW SPEEDS

Investigation of the test limitations due to flow breakdown. Work at RAE (14.2) on interference effects for models with lift jets in closed tunnels was referred to in AGARD-AR-83 and the report on this will soon be issued. The corresponding tests in open working-section tunnels have been cancelled. Work at Surrey University, under RAE sponsorship, on mathematical modelling of jets to provide a basis for jet interference calculations has now started, but no results are yet available.

A short paper[1] has summarized comparative data obtained in the HSA 4.6 m x 4.6 m tunnel (42.2) and the NRC (Canada) 9.2 m x 9.2 m tunnel on a model with sixteen lifting fans. The work described forms the first part of a programme designed to assess tunnel interference effects for such configurations, and is concerned with identifying the boundary condition of *incipient stagnation* which can be used to establish a minimum tunnel operating speed. The data collapse very well if the ratio of height above the floor to effective fan exit diameter is plotted against the ratio of free-stream to fan exit jet velocity, and appear to be independent of jet velocity, jet inclination, number and disposition of jets and jet shape and size. Inter-tunnel correlation using these parameters is not good, however, indicating that some as yet unidentified tunnel characteristic must be significant. Full reports on this work are expected to be published in the near future.

Some work has been done at Westland Helicopters Ltd[2] to investigate the possibility of extending testing limitations due to flow breakdown by removing some wall panels from the otherwise closed test section of a 3 m x 3.6 m low speed tunnel. Tests made on a four bladed 1.8 m diameter rotor indicated that such an arrangement enabled at least qualitative testing to be done on hitherto oversized rotors, though no attempt was made to evaluate interference effects for the vented configuration. The work, which has not been reported directly to this Working Group, falls within the scope of one of the recommendations made in AGARD-AR-83.

Wall corrections and limits of applicability. Work has continued at DFVLR (42.1) and (42.4) to check the validity of windtunnel correction procedures. Evaluation of experimental data obtained on two aircraft models in five low speed windtunnels (42.1) has been completed and good agreement between results from the different tunnels was found for one model after corrections had been applied (corrections of Kraemer for the open test section and of Kraemer & Vayssaire for the closed test section). No details of the work are generally available at the present time.

The DFVLR (PW) study (42.4) on displacement corrections and their limits of applicability has now been reported in English.[3,4] In this case correction methods cease to be valid when the dynamic pressure varies significantly over the different parts of the model surface, and attention is drawn to the factors which have an important influence on this variation.

New work has been reported[5] at ITS (42.8) on the controversial subject of the effects of wake blockage when testing high-lift models in low speed windtunnels. The method presented has been used for several years in windtunnel tests with thrust reversers where blockage effects are large.

Work continues at Washington University (42.5) on the evaluation of existing theories for wall corrections. Tests on a model aircraft have begun in the 2.4 m x 3.6 m windtunnel, both with and without a 1.2 m x 1.8 m insert. The model has a 0.9 m span non-swept wing with a symmetrical profile and a tail that can be mounted in either a low or a high position. It is mounted on a six component external balance and has two jet lift engines mounted separately (off the balance) and located near the fuselage forward of the wing. No results are yet available.

At VKI (42.6) work has continued on the extension of Joppa's vortex-lattice theory to an open working-section tunnel. The lifting wing is represented by a single horseshoe vortex, and a computer program has been written which allows for the effects of wake relocation. The program has been run for a number of cases with rather moderate values of C_L/A, to allow comparison with some available experimental data, and the results reported in a student project report.[6] Agreement between predicted and experimental values of interference factor is reasonable but inconclusive, and some comprehensive experimental investigations of the flow field associated with a high-lift wing are needed to validate the theory.

The study at FFA (42.7) using vortex lattice methods has been completed and a report[7] published.

The design of slotted or porous test sections. Experimental work at AEDC (43.1) referred to in AGARD-AR-83 has been reported.[8] Theoretical work to develop a vortex-lattice method for the computation of interference in a slotted wall tunnel for V/STOL type models has been continued. A scheme in which a wing/centrebody model and the slotted walls are represented by appropriate vortex-lattice configurations is currently being brought into use.

At UBC[9,10] (43.3) two-dimensional tests have been conducted on a range of sizes of aerofoils of three different profiles and good agreement obtained with potential-flow thick aerofoil theory. It appears that uncorrected C_L values and C_p distributions, accurate to within 1%, can be obtained for a wide range of aerofoil shapes, sizes, and lift coefficients, using a solid wall opposite the aerofoil pressure side and a slotted wall with 60% open-area ratio opposite the aerofoil suction side. Development work continues.

A new theoretical and experimental program at NLR (43.4) is designed to verify a new method for calculating wall interference in ventilated test sections of finite length. This method has been previously reported for two-dimensional flows (54.1), and (5 10.2), and has now been extended to three dimensions.[11] It is intended to study the possibilities of eliminating at least the variation of the wall-induced velocities over the model and of predicting the remaining wall corrections in a reliable way. The experimental work will be conducted in collaboration with DFVLR.

Techniques for two-dimensional and half-model testing. Progress under this heading has been made at FFA[12] (46.3), where the study of half-model high-lift techniques for the 3.6 m diameter tunnel is complemented by the theoretical work referred to earlier (42.7). A number of wings with sweep angles 0° – 35° have been tested in combination with half-fuselages, using an insert with porous boundary layer suction on the reflection wall. Comparisons with results obtained on a full model test in the 5 m x 7 m tunnel in Eidg. Flugzeugwerk, Emmen, Switzerland, indicate that the half model testing technique is of great value in the development of high-lift configurations.

Conclusions and Recommendations. Work to investigate flow breakdown continues, with the object of determining criteria which can be used to assess testing limits, but there has not been much progress in this area since AGARD-AR-83 was issued. The new results reported indicate that the understanding of the phenomena involved with multi-jet configurations in closed windtunnels cannot yet be considered satisfactory. It is important, therefore, that the rather small number of investigations currently in progress should be continued, and it is desirable that their number should be increased. It is encouraging to note that a start has been made on the investigation of the alleviating effects obtainable from ventilated walls, though the results so far reported are of a preliminary nature.

A substantial effort is currently being made to investigate wall corrections and theoretical methods for their estimation. It would appear that most of the factors likely to be significant are in fact being considered in one or other of the investigations, and this now includes the important problem of wake recirculation in the windtunnel. Systematic studies are being made to assess the validity of existing methods for the prediction of wall corrections; results so far have been rather inconclusive, but jobs in progress promise to effect considerable clarification and should eventually indicate the respects in which these methods need to be improved.

There is no progress to report on the use of self-correcting windtunnels for high-lift low-speed testing, but there is considerable interest in the use of ventilated walls to achieve smaller wall corrections. Some new experimental results confirm the usefulness of the approach, and further experimental work is planned. In addition, theoretical work is proceeding to assess the validity of methods for the mathematical simulation of ventilated test sections.

Finally, some new results are available to confirm the value of half-model testing with boundary layer control employed on the reflection wall. It is recommended that application of this technique be pursued.

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5. SPECIAL PROBLEMS OF TESTING AT TRANSONIC SPEEDS

The main thrust in this vital area is still centered on wall interference effects on the model and the actual behaviour of the transonic wall in existing facilities. Significant efforts are noted in the application of numerical methods for treating the flow field around a model in a transonic windtunnel. The adaptive wall concept is further advanced and a number of new jobs have emerged here. Progress is reported on noise generated by ventilated walls. No further efforts are reported on spurious scale effects due to heat transfer. However, a study on heat transfer effect on shock/boundary layer interaction is noted. Rather remarkable is that under subsection 5.14 "Design of Plenum Chamber" not a single activity has been reported.

Windtunnel wall interference. The two-dimensional case is still attracting substantial interest. The work at NAE and ONERA has continued. The use of wall pressure measurements in combination with subsonic theory to determine the appropriate porosity factors for the floor and ceiling is now applied on a routine basis at NAE (51.8). The unequal porosity parameters for the floor and ceiling appearing for lifting models (even though the geometric porosity is the same), results in significant blockage effect due to lift.[1]

In further work at ONERA (51.9) along the same lines, wall pressure measurements obtained in the R1 Ch and S3MA windtunnels are used in combination with a linear method or the transonic small perturbation method to determine the porosity characteristics. Also, work is under way at ONERA (51.7) using an analytic wall correction method to establish an asymmetric porosity configuration (unequal floor and ceiling porosity) that will yield negligible wall corrections for the S3MA windtunnel.[2]

Much experimental work has centered around models of the classical NACA 0012 profile. The influence of the sidewall boundary layers on results for models of the NACA 0012 profile and the ONERA LC100D profile was investigated in the R1 Ch (51.5) windtunnel. The thinning of the boundary layers was effected by suction upstream of the model. No simple method to correct results for the presence of the sidewall boundary layers could be established; making the boundary layers as thin as possible, is the recommended approach.[3] Similar work has also been carried out at IMFL (51.6) with different types of porous material.[4,5]

No results have been reported on the investigations of three models of different size of the NACA 0012 profile at DFVLR (BF) (51.2) to determine blockage corrections.

Convair (51.13), in experiments performed in the AFFDL 15" two-dimensional slotted windtunnel, has used measured wall pressures in combination with the unsteady finite difference procedure to determine wall corrections.[6]

LRC (5 10.8) has also established the usefulness of using measured wall pressures for a slotted windtunnel as boundary conditions for assessing wall interference in two-dimensional subsonic flow.

At AFFDL (5 10.6) a series of two-dimensional tests in the trisonic gasdynamic facility has demonstrated the utility of thickness contouring slotted walls. The upwash interference over the first 60% of chord could be eliminated in this way.[7]

Work with *three-dimensional models* deals with bodies of revolution, half (or reflection plane)-models and full models.

At LaRC (51.12) drag measurements have been obtained near $M = 1$ in the 16 ft transonic windtunnel for a series of bodies of revolution for comparison with flight test data. The bodies were geometrically similar yielding blockage ratios from 0.00044 to 0.00017. In spite of the very low blockage ratio it was found that all models had a lower drag rise Mach number than the free flight body, also that the smaller the model, the lower the drag rise Mach number. Since change of model size meant a change in both Reynolds number and blockage ratio, work is now underway to separate the two effects before any definite conclusions can be drawn.

Half model technique has been employed by Ames and FFA but for entirely different purposes. The Ames (51.11) investigation concerns comparison of results for an RAE model tested both in the RAE 8 ft and Ames 11 ft transonic windtunnels. This comparison is to form the basis for determining experimentally the constants in the generalized homogeneous boundary conditions, using the solution by E.M.Kraft, slightly modified, for lift interference in a windtunnel with ventilated top and bottom walls. The analysis is being extended to all four walls ventilated.

The work at FFA (51.5) was aimed at establishing the usefulness of the half model technique at transonic speeds. Components from a full swept wing model (1/25 scale), tested in the NAE 5 ft windtunnel, were used for a half model that was investigated in the FFA TVM 500 windtunnel. The two sets of data show good agreement when compared at same Reynolds number. A larger (1/8 scale) half model, geometrically similar to the one above, was also tested in the NAE 5 ft windtunnel in order to obtain data at higher Reynolds number. Effects of Reynolds number on drag could be seen up to the highest Reynolds number tested ($Re \sim 15 \times 10^6$).[8,9,10]

The extensive program with the ONERA calibration models being tested in a number of transonic windtunnels seems to have run its course, apart from tests still to be conducted in the NLR HST. Tests have been completed in the following windtunnels: ONERA S2MA, S3MA, S3Ch, IASC Sigma 4, DFVLR 1 m, FFA S4 HT and TVM, ARC 11 ft, AEDC 4T and 16T, RAE 8 ft and NAE 5 ft. The purpose of these tests has been principally to establish suitable wall correction methods for realistic aircraft configurations through the use of identical models in various size windtunnels. Much analysis of acquired data still remains, although ONERA and FFA must be credited with accomplished analysis and reporting.[11,12,13] The working group on transonic test section design considers the ONERA models to be especially Reynolds number sensitive and recommend a new standard model for studies of wall interference effects. See Appendix 6.

The effective porosities of the ONERA S2MA and S3MA windtunnels have been determined with the aid of the calibration model data (53.3). Corresponding wall corrections are now employed in industrial testing as function of span and angle of sweep. The confidence in these corrections is such that models with a span of up to 80% of tunnel width can be tested. An investigation has also been carried out to determine test section conditions related to model scale that would yield wall corrections within the normal scatter of data.

LeRC (5 13.7) has completed an investigation in the Mach number regime 0.6 to 1 in the 8 ft x 6 ft windtunnel on a series of geometrically similar winged-body configurations (not the ONERA models!) representing blockage ratios from 0.1% to 2%. Measurements included fuselage pressure distributions. The effect of blockage was found small up to $M = 0.95$. The effects of variations in local wall porosities and sidewall contour was investigated. The porosity changes were found effective in reducing wall disturbances up to $M = 0.975$ but not so the sidewall contouring. Model support interference effects were also investigated and in addition to normal sting mounting, wing-tip mounting and fuselage forward swept support strut mounting were investigated.[14]

A few investigations have been concerned principally with the characteristics of the ventilated wall. Significant progress is reported by FFA (59.1) on the investigations of the flow in a slotted wall. A special pressure probe traversing along and across the slot has been designed and used. The measurements have confirmed a tentative flow model for the slotted wall and computed pressure differences across the wall show good agreement with experiments.[15,16,17] Furthermore, a fully three-dimensional inviscid theory for the wall interference in a slotted wall windtunnel has been developed.[18]

At AEDC (5 10.4) the wall characteristics of the 1 ft transonic windtunnel has been determined experimentally. Using an inverse transonic potential flow program the flow angle distribution at the wall is then calculated. The results are applied to calculations of mass flow distribution and boundary layer growth at the wall using the Potanker/Whitfield computer program. Agreement between calculated and measured boundary layer thickness at the test section exit indicates overall consistency of the approach. Furthermore, the concept of axial variation of wall resistance to reduce

interference has been analytically demonstrated for a finite airfoil[19] and techniques are available to calculate the interference effects for an arbitrary distribution of porosity.[20]

The development and *application of numerical methods* have further advanced.

The NLR panel method (54.1), using the non-homogeneous boundary condition at the ventilated wall and taking into account the finite length of the test section, has been applied to a study of two-dimensional flow. This study revealed that the lift interference could be significantly reduced, without affecting blockage, if the upper and lower plenum chambers were not interconnected. The method is being further refined by applying measured wall pressure distributions as boundary conditions.[21,22]

An extension of the NLR panel method (5 10.2) to three-dimensional flow in a ventilated wall windtunnel has been accomplished.[23]

The transonic small perturbation method has been applied to two-dimensional flow both at ONERA and RAE. ONERA (59.2) has found this method too restrictive and is not pursuing this avenue. At RAE (59.4), the method has been applied to slotted and solid wall windtunnels. Experiments in the RAD 8 ft x 6 ft windtunnel agree with calculations, although difficulties are encountered in determining the correct P-value for slotted walls. Calculated results show that the interference effects in a slotted wall windtunnel is much more severe for a supercritical airfoil than for a classical one. Calculations also confirm that the linear subsonic theory may be used to provide adequate wall corrections in a perforated wall windtunnel for lift and pitching moment up to $M = 0.8$. However, it is not considered adequate to provide blockage corrections in transonic flows.[24]

At AMDBA two approaches for wall corrections in three-dimensional transonic flow have been investigated; an analytical method and the vortex lattice method. It is now reported (52.1, 5 10.7) that the vortex lattice method is the favored approach and in current use.[25,26]

Work has also been underway for some time at AMDBA (53.4) to arrive at a method for computing the choking Mach number in a solid wall windtunnel with a model generating high lift and drag. It is now reported that such a method exists and some verifying tests are scheduled.

Oceanics (59.3) has developed a theoretical method for M close to one. It is based on local linearization together with an integral method for treating the flow at M close to and equal to one. Their study reveals, that in a perforated wall windtunnel at $M = 1$, thick models experience less interference than thin ones. Experimental data obtained in a gas-dynamic-hydraulic analogy facility confirm this prediction. A new, simplified method for calculating lift on thick wings and airfoils in unsteady flight at $M = 1$ has also been developed.[27]

At the University of Arizona work has been started on the development of rapid methods for calculation of steady and unsteady transonic flow with emphasis on wall interference effects (59.5).

There is nothing further to report on the time dependent numerical procedure for solving inviscid transonic flow, developed at VKI and University of Liege (5 10.1). The method was successfully tested in simple examples.

NAE (51.8) has developed an influence function method for computing wall effects on single and multicomponent airfoils, cascades, and vortex roll-up in a solid or ventilated wall windtunnel.[28,29,30]

AEDC (5 10.3) has developed an integral technique for solving the nonlinear transonic equation and applied this to nonlifting airfoils in a two-dimensional perforated wall windtunnel. The integral approach yields an order of magnitude reduction in computing time over other methods.[31] It is demonstrated that transonic interference effects are model dependent; for example, the porosity required for zero blockage is more a function of thickness distribution than blockage in the classical sense.[32] Furthermore, the Newman-Klunker computer program for calculating three-dimensional transonic flow about a model with arbitrary windtunnel boundary conditions has been adapted to the AEDC computing facilities.

Effective computational methods are being developed at LaRC (5 10.8) for treating two- and three-dimensional flow in slotted and perforated wall windtunnels. Improved theoretical boundary conditions have been developed for slotted walls and are being extended to alternately staggered rod walls. The wall-induced perturbation field has been defined rigorously within the context of three-dimensional transonic flow computations along with a criterion for assessing the correctability of windtunnel data to free air conditions.[33,34,35]

ATL (5 11.3), which previously reported on the development of a new and rapid technique for solving the three-dimensional non-linear small disturbance transonic equation, has carried out further theoretical studies based on more exact computer codes. The importance of correcting for the finite length of the test section has been established.[36]

Studies on *new wall concepts* are progessing at various places. At Calspan (5 11.2) tests have been completed in their 1 ft windtunnel with adaptive porous walls (based on Sears' proposal[37]) on a 6 inch chord two-dimensional model

of the NACA 0012 airfoil. The test Mach numbers were 0.55, 0.65, and 0.725. Only a small number of iterations were found necessary in order to obtain low interference flow. Data also show that high interference tests do not correspond to any pseudo-angle of attack or Mach number.[38,39] Tests are continuing at $M = 0.85$ and 0.9.

At ONERA (5 11.4) the adaptive flexible solid wall concept has been successfully applied to two-dimensional tests in the S4Ch windtunnel. Again, a model of the NACA 0012 was the test object.[40] Application of this concept for the three-dimensional case is also under study.

At AFFDL (5 10.6) a nine-inch rod wall transonic test section is being designed for studies of adaptive wall techniques with the rod wall.

USAA (5 11.5) reports on the development of a subsonic two-dimensional flexible wall test section. Tests on a cylinder with 29% blockage at subcritical Reynolds number have demonstrated the achievement of interference free flow Initial testing of an airfoil model (NACA 0012-64 profile) is complete and further testing is aimed at gaining experience with the adaptive wall technique at angles of attack through stall. Improvements are sought of increasing angles of attack through stall. Improvements are sought of increasing the rate of convergence of the walls from straight to streamlined contours.[41,42]

The ARC investigations (56.2) with wedge shaped walls in their 2 ft x 2 ft transonic windtunnel has proven inconclusive due to severe boundary layer build-up. Future plans are to reconfigure the test section to facilitate testing of new wall geometries.

There is no progress to report about the planned experiment at USAA (5 11.6) to explore the possibility of attenuating the reflection of shock- and expansion waves against a solid contoured wall.

Similar work is being pursued at the University of Stuttgart (5 11.8), although the walls are elastic. Theoretical work has progressed to report stage, but the experimental part of the study is still pending.

Both IMFL (57.2) and AMDBA (57.3) are investigating the use of homogeneous porous material for the transonic windtunnel wall with regard to shock wave cancellation in slightly supersonic flow.[43] Another approach followed at GASL and reported in Reference 44 is a "land and groove" wall geometry with variable porosity. Experimental results show that the wall reduces the reflected shock strength. However, the required porosity distribution was found to be a sensitive function of Mach number in the low supersonic range, with Mach number accuracies of 0.01 or better required to obtain repeatable results.

At AEDC (5 11.7) a computer simulation technique has been developed, that models the flow in a two-dimensional test section in conjunction with a numerical solution of the exterior flow. The relationship between flow variables that must be satisfied to ensure unconfined flow can thus be established. The technique has been used to study the effect of probe locations and the required measurement accuracy to ensure convergence. In order to reduce computing time an integral equation method is being developed for the external flow field. Furthermore, a three-dimensional numerical simulation of the adaptive wall is being developed.

FFA (5 11.9) is examining the feasibility of a new wall concept that can best be described as a convertible wall. With the same wall components the wall configuration can be changed from slotted to perforated walls with variable porosity and vice versa. A case is studied where perforations are limited to four wide slots (40% of wall area), the perforated plates have slanted holes and the porosity can be varied up to 20%. Preliminary windtunnel tests with a cone cylinder in the TVM 500 windtunnel have shown promising results.

Finally, in this context, the correctable interference tunnel concept, being considered at LaRC should be mentioned.[45] Limited adaptive wall control would be used to reduce interference to analytically correctable levels. Pressure measurements at the walls would provide the boundary conditions for calculating the remaining interference, using an analysis procedure, that is rigorously applicable to transonic speeds.

Noise generated by ventilated walls is intimately connected with the flow quality in the test section. Up till recently the question of flow quality has defied many attempts at clarification. However, a very important document on the subject has now been issued by AGARD[46] that defines the flow quality requirements, for practical purposes, for transonic windtunnels with short run time. "This report should be mandatory reading for all those concerned with the great variety of tests to be done at transonic speeds, whether they design the experiments or carry them out and evaluate them." (D.Kuchemann).

Several investigators report on successful schemes for reducing noise generated by ventilated walls. ULC (5 12.1) found that positioning a plate in the plenum chamber parallel to the perforated wall could give substantial reduction in noise level under certain conditions. ULC is also studying the characteristics of perforated walls covered with gauze. In addition to studying the acoustic level in the working section, the boundary layer development and the shock cancellation characteristics of the composite wall are being investigated.

Both ONERA (5 12.2) and AEDC (5 12.5) report that the application of fine mesh screen or gauze over the perforated wall can reduce the noise level to that of solid walls. At ONERA fluctuating pressure and turbulence measurements were carried out in the S2MA and S3MA windtunnels using the AEDC 10° cone.[47,48] AEDC also confirms that longitudinal splitter plates in 6% – 60° inclined holes are an equally effective means of reducing noise without affecting the shock wave cancellation or subsonic wall interference characteristics of the wall. However, the same modification to a variable porosity 60° inclined hole wall, while substantially reducing the noise level, virtually destroyed the wall's wave cancellation properties. AEDC is also looking at the effect of screens on the noise produced by other wall geometrics.[49] An investigation of the acoustic characteristics of the rod wall is also reported by AEDC.[50]

Progress on the development of a theoretical method to calculate the noise generated by a ventilated wall is reported by Nielson (5 12.4). The theory, which is based on the stability of slightly non-parallel shear flow, has been brought to a practical stage of development and computer programs have been generated. Computations have yielded the most amplified frequency for a given flow condition. The calculations indicate an edgetone frequency in the correct order of magnitude range for the AEDC 16T windtunnel.

The effect of heat transfer on test results. The only investigation so far under this heading is the one previously reported by RAE (5 15.1). It was noted that spurious scale effects could occur due to heat transfer.[51] However, it is indicated in (56.3) that tests are being carried out at Calspan to investigate the effect of heat transfer on shock/boundary layer interactions.

Conclusions and recommendations. Since the "MiniLaWs" activities started, significant progress has been made in many of the areas discussed in this section. A better understanding of the wall interference problem in existing facilities has definitely emerged. This better understanding is primarily the result of the combined effort that has gone into the development of analytical and numerical tools for treating the flow in a ventilated windtunnel and the experimental work that has been directed towards the understanding of the flow characteristics of the ventilated wall itself. In many two-dimensional facilities it is now common practice to measure the pressure distributions on the ventilated top- and bottom-walls for defining the appropriate wall boundary conditions to be used in the method applied for calculating the wall interference effects.

Several such methods have been developed and are applied in practice; subsonic linear theory (NAE, RAE), the NLR panel method (NLR), transonic small perturbations (RAE), an inverse transonic potential flow method (AEDC), unsteady finite differences procedure (Convair). Although some limited assessment of the "range of applicability" of some of the methods has been made, it is recommended that a systematic study of the merits and applicablity of these methods be carried out, including establishing the limits (e.g., C_L, M, C/H, P) for when results are correctable or not.

The above more or less holds for the three-dimensional case as well, although it is less clear which methods for calculating wall interference have reached a practical stage of development. It is gratifying to note however, that, based on the ONERA calibration model program, ONERA has established a correction procedure for the S2MA and S3MA windtunnels that is applied on a routine basis. Much of the work with the ONERA models is still only reported as "has been conducted" and it is urged that the analysis and subsequent reporting of results be speeded up as much as possible.

Encouraging results have been reported from experiments with the adaptive wall technique, using porous or solid walls, in two-dimensional windtunnels. Other adaptive wall concepts are also being studied; e.g., rod wall and elastic wall, but little information is available on the progress of these studies. And, as pointed out in the previous MiniLaWs report, the road to the three-dimensional adaptive wall is long. However, a computer simulation study on such a concept has been initiated. A word of caution may here be in place. The success with the two-dimensional adaptive wall has primarily been based on experiments with the NACA 0012 profile. However, as reported in one study, a supercritical airfoil is much more sensitive to wall interference than a classical one. Further exploratory work with the adaptive two-dimensional wall should therefore include some supercritical airfoil model, so that the practical problems associated with more sensitive models can be assessed.

In Appendix 6 the working group on transonic test section design recommends the development of a standard model and suggests a standard test procedure for evaluating interference correction methods and new test section design concepts. The AGARD FDP should recommend these tasks be undertaken by a specific single agency. Advice for extensions of present day transonic tunnel calibration methods is offered in Appendix 6.

Effective means for suppresssing the noise generated by perforated walls are now at hand. Splitter plates or gauze have been found equally effective in reducing the noise level of a perforated wall to that of a solid wall. However, the question of whether or not the wall generated noise has an influence on aerodynamic measurements, is still open to debate. Also, further work is required to establish the interference characteristics of the "quiet" walls, before they can be considered to be a viable alternative.

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6. FLUID-MOTION PROBLEMS

Most of the material presented in this section is concerned with windtunnel testing, primarily with the influence of the boundary layer, but also with the need for high Reynolds number simulation or testing. Since most of the reported research is on viscous flows, some of it is covered by the activities of Eurovisc (European Research Programme on Viscous Flows). The Eurovisc Annual Report 1976 (Ref.1) has recently been issued and many references are made here to specific chapters on items in that report, which may be read in conjunction with the present report, and which gives additional information.

The effects of different flow disturbances and surface imperfections on boundary layers, including the mechanism of transition. The results from the experiments in the NLR Pilot Tunnel on the influence of artificially-generated sound disturbances on flow separation (62.1) were reported at the FDP London Symposium.[2] It was confirmed that, on this comparatively low noise level, no influence of an increase from 0.35% to 0.6% for CP (rms) was found on separation or on the lift of a supercritical aerofoil. Since this is the range of noise levels discussed for new transonic windtunnels, the results are of great interest.[3,4]

The earlier reported work at RAE (B) (62.2) which also dealt with the response of a turbulent boundary layer to acoustic excitation and with similar conclusions awaits the final analysis and reporting.

Three US investigations are reported. The first, at AFFDL (62.3), has the objective of obtaining the transition-point in free flight on a 10° cone in the transonic regime, and to compare this with windtunnel results. The preparations for the flight tests are still going on. When completed and analyzed, they can be expected to give valuable information. The second investigation, at AEDC (62.4), is concerned with the correlation of transition Reynolds number with noise and turbulence levels in transonic tunnels in US and Europe. An analytical model[5] for prediction of the onset of transition has been derived and gives satisfactory results. It is now reported that the correlation of the windtunnel data indicates a non-monotonic variation of transition Reynolds number with Mach number. A flight test of the model is now also planned, which intends to cover the windtunnel test conditions.

The third US investigation, at AEDC (62.5), concerns the effect on a turbulent boundary layer from varying free-stream acoustic levels and free-stream velocity disturbances, the latter introduced by placing a lattice of steel rods in the stilling chamber. Provile measurements of the turbulent and mean flow have been made with both a split film and a hot wire anemometer. Reduction of the data is under way. Reference should also be made here to Section 5 of this report, where the influence of noise generated by ventilated walls is discussed.

On the general problems of possible mechanisms for transition in boundary layers, which are of great relevance to TES objectives, reference is again made here to the Eurovisc Working party on Transition in Boundary Layers.[1] The work of the group and the work on transition in the USA is reflected in the report of the conveners of this subcommittee on Laminar-Turbulent Transition in Boundary Layers (Appendix 7). Much work, both theoretical and experimental, is needed to provide a rational prediction procedure for boundary layer transition.

Results from the earlier investigation on transition on the AEDC cone at RAE (B) (61.3) have now been published.[6] The AEDC cone has also been tested by ONERA and the results were reported at the London Symposium.[7] Further transition tests on a new 10° cone in various European facilities are being actively considered. The cone would be manufactured and instrumented by NLR and tested in some of the facilities used for the AEDC cone.

The investigation at RAE (B) (63.1) of the effect of surface imperfections on boundary layers, earlier reported,[8] has been extended to measurements of fluctuating pressures upstream and downstream of a square ridge. A new investigation at UTSI (63.3) concerns determination of the effects of pressure orifice on skin friction and turbulent boundary layer characteristics.

From the investigation at ULICA (67.1), where turbulence measurements in and behind the reattachment region of a backward facing step earlier have been reported[9] are now completed and written up in thesis form.[10]

The work at NAE (61.5) concerning skin friction on two dimensional aerofoils with different roughnesses using the razor-blade technique awaits calibration by means of a skin friction balance. This calibration was expected to be performed during 1976. FFA reports a new investigation (61.6). The effects of different transition trips on the downstream behaviour of the boundary layer on a flat plate have been investigated at low speeds using a single DISA hot wire to record mean velocity and streamwise fluctuating velocity profiles as well as spectra. The results indicate that, at these rather low Re, the transition region before fully turbulent boundary layer behaviour is obtained, is quite long. A detailed description of the test setup and the results are published.[11]

The extensive investigation (6 14.1) at NAE in cooperation with Laval University on the experimental techniques for measuring turbulent skin friction proceeds and the floating element is further developed. More reports[12,13,14] have been published.

Finally, it should be pointed out that many more investigations of both direct and indirect interest to this field are reported in the Eurovisc Annual Report (1976) (Ref.1). The following chapters especially contain much relevant information: Chapter 2: "Transition and Reversed Transition", Chapter 10: "Separation and Reattachment", Chapter 14: "Pressure Fluctuations, Aerodynamic Noise Generation and the Effects of Free Stream Fluctuations", and also Chapter 15: "Excrescences and Roughness Effects".

On the subject of the influence of turbulent boundary layers due to disturbances, three recent publications should be mentioned which concern the influence due to changes in the free-stream turbulence.[15,16,17]

Techniques for simulating flows at higher Reynolds numbers and comparison between results in the laboratory and in flight. A study at RAE (B) (64.3) of transonic scale effects on swept wings is now completed and is being reported. At RAE (F) (64.4) the results earlier obtained from a transonic investigation into the effects of compressibility at high-lift, low speed has partially been analyzed. It is concluded that further investigations are needed and consideration is therefore being given to a model to be tested in the new RAE 5 m tunnel.

The experimental study at ONERA (64.5) on simulating higher Reynolds numbers by using enlarged leading or trailing edges is now considered completed as regards the trailing edge studies. These results have earlier been reported.[18] The study of the flow around the leading edge continues and a new report[19] is announced. The possibility of simulating higher Reynolds numbers by means of surface roughness is further pursued at ONERA (64.12), now with some experimental tests. The beginning of transition is shown to take place at the position of roughness; however, the roughness is not effective downstream and restrictions are also necessary as regards applicability at angles of attack.

The experimental investigation at LeRC (64.6) on Reynolds number effects on boattail pressure drag has been completed and the analytical effort is continuing. Data indicate a strong sensitivity of boattail pressure drag to approach boundary layer thickness when extensive regions of separated flow exist on the boattail. In these cases, both data and analysis show a decreasing drag with increasing approach boundary layer thickness. (See also Section 3).

More results from the work at OSU (64.7) is now reported. The investigation concerned high Reynolds number (over 3.10^8 per meter) transonic aerofoil and transonic wall-interference problems.

Interference is determined by comparing surface pressure distribution on the two-dimensional model with Krupp-Murman calculations at subsonic speeds and with data from tests on a 6-inch chord model in the 8 ft Calspan tunnel at transonic speeds. Pressure differences between the upper plenum and the lower plenum is significant and may lead to an understanding of the results. The investigation will continue. The joint investigation between RAE and Ames Research Center (64.9) on scale effects on the transonic flow on swept wings has been completed and analysis of results for wall-interference effects is in progress.

The comprehensive research programme at ONERA (6 13.3) concerning the correlation between windtunnel results and theoretical predictions and flight tests continues. Fair agreement is reported between flight and windtunnel tests on Mirage III between M = 0.7 and M = 1.85 and the earlier planned flight tests on Nord 2501 have now been performed. The results obtained from boundary layer transition location versus angle of attack have been published[20] and windtunnel tests on the Nord 2501 full scale leading edge used in the flight tests will soon be tested in the Toulouse S10 windtunnel by using the enlarged-leading-edge-method. Free flight Reynolds numbers will be obtained. This comparison between free flight and windtunnel tests will be looked forward to with great interest.

The high Reynolds number testing project at Ames Research Center (6 13.7) has made further progress. The windtunnel tests with the C-141A semispan model are completed and reported.[21] Design of wing-fore-body model for test in MSFC Ludwieg Tube Tunnel is in progress.

A new contribution comes from AFFDL (6 13.8). Its ambitious objective is to flight demonstrate the performance improvement of a supercritical wing and to obtain flight test data for correlation of analysis and windtunnel results for assessing the fidelity of tunnel simulation and for developing new windtunnel modeling and testing techniques. Flight test data will be obtained for Mach numbers up to 2.2 with emphasis on transonics. Windtunnel tests will be conducted on full span and half span rigid models and flexible pressure models which are aeroelastically similar to the SCW flight vehicle.

The progress report gives many interesting results and will therefore be included here in some detail: the TACT (Transonic Aircraft Technology) program involves a detailed correlation of windtunnel and flight test determined aerodynamic forces. The TACT force accounting system establishes a reference engine configuration and all aerodynamic forces that vary with power setting are included in the thrust determination model. The thrust dependent aerodynamic forces are determined from windtunnel tests performed parametrically about the engine reference conditions and sufficient data are acquired in flight to verify the ground test results. Drag is a ground/flight test correlation parameter. The variation of forces with power setting exists on other fighters but did not affect the calculation of handbook performance since flight test derived drag and thrust were combined to obtain the performance and separation of the forces into correlative values of thrust and drag was not required. AFFTC is vitally interested in the TACT program and the determination of correlative values of drag in that it would reduce the amount of test time required for any new airplane. Progress to date has reduced the variation of flight test drag with power setting to plus or minus 10 counts. Flight tests of the F-111 with the supercritical wing are in progress. Windtunnel tests of 1/2-scale models that duplicate the flight airplane are under way at Ames Research Center. Nozzle afterbody windtunnel tests for the program have been completed at Langley Research Center.

The inlet performance testing criteria being studied at AEDC (6 13.5) have been completed, a final report[22] is issued and an AIAA paper[23] presented.

DFVLR (B) (6 13.1) is carrying out thrust and jet-flow measurements under laboratory and flight conditions. The tests in the static test bed are completed and the first flight results are expected at the end of 1976. This is the only contribution in this important area. A publication is reported.[24]

The investigation at RAE (B) (6 13.2), which aimed at establishing the scaling laws for the intensity of buffeting and also at investigating Reynolds number effects up to full scale value continues.

Half-model tests in the ECT and in the 8 ft x 8 ft tunnel are completed and results are being compared with results of flight tests. Complete model tests in the latter are planned for 1977.

Separation in three-dimensional flows; conditions and consequences. At the NLR (65.2) the detailed flow investigation of shockwave/boundary layer interaction has continued and data reduction and analysis are in progress. Two publications[25,26] have been issued. A similar investigation at AEDC (65.5) reports new activities. The flow field over an asymmetric bump has been computed by combining a boundary layer method with an inviscid transonic flow solution. Laser velocimeter measurements were made of the flow field including the region inside the shock/boundary-layer interaction. A new report[27] is available. (Progress report on the study at NC State (65.4) is missing.)

The next investigations all treat the influence of the variation of Reynolds number, in some cases up to very high values. At LaRC (65.6) the effects of changing the wall porosity is also included. It is found that at supercritical conditions, wall porosity had a large effect on the airfoil pressure distributions and shock locations. Standard linearized theory was generally inadequate to account for the wall interference effects on angle of attack and Mach number.

Three new publications are reported. At CAL (65.8) the detailed investigation[28] of shock wave/boundary-layer interaction, with its important implications regarding full-scale simulation, has progressed with further tests. See also (56.3). At UTSI (65.9) new results are reported on the investigation of turbulent-boundary-layer separation up to very high Reynolds numbers (150 millions). The experiments at subsonic velocities have been completed and are under way for low supersonic Mach numbers. Two new publications[29,30] are reported. In the earlier investigation at DFVLR (G) (67.3) on the reattachment and subsequent trailing-edge separation of a shock-induced separation a new boundary layer probe is manufactured. The test program is under way. An earlier report[31] is available. For related investigation at RAE (B) see also Eurovisc Annual Report[1] 1976, Job 9.2, where a detailed experimental study of the shock wave/

boundary-layer region is reported. In the same Eurovisc Report is also reported a detailed experimental analysis of shock wave turbulent boundary layer interaction at transonic velocities using laser anemometry and holographic interferometry (Job 9.4).

The work at NLR (66.1) on a programme[32] for transonic buffeting research progress in a second phase with measurements of pressure-fluctuations in and upstream of the shock-induced separation on a rigid two-dimensional supercritical profile. The data are now being analyzed. The study of various cases of the consequences of separation in three-dimensional flows at NAE (66.2) is now completed and reported.[33,34]

Wakes and jets. The first four items are from NLR. In (6 11.1), which concerned the effect of some jet parameters on the thrust-minus-drag of an axisymmetrical body at transonic speeds, is completed and gives now a new reference.[35] (6 11.2), not reported earlier, describes a method developed for the prediction of the flow field around air-frame-jet combinations. Agreement between calculated and measured pressure distributions on the wing is good. Many references are given, for instance 35, 37, 38. This investigation, and also (6 11.3) and (6 11.4) are thought to be of only marginal direct interest here; their main importance should be for aircraft designers. However, they are included here since the results may lead to the identification of scale effects, which should be simulated in windtunnel tests and also since the calculation methods may be used when calculating the interference between windtunnel walls and free-jets. No new results are however reported this year.

The investigation at FFA (6 11.5) concerned an experimental and analytical programme[39,40] on the effects on the afterbody and near-wake environment of strong interactions between a central propulsive jet and a supersonic external stream. Tests at angles of attack are going on and also determination of the influence of control surfaces. The possibility of using cold flow to simulate hot flow jets by means of a new plume modelling law has been investigated.

From the investigation at RAE (F) (6 11.6) regarding afterbody drag at transonic and supersonic speeds has progressed. The new results show that the effects of mutual interference are important in practical configurations and suggest possible methods of predicting the total drag from part-body tests. The investigation is proceeding.

The ARA (6 11.7) has now successfully completed the programme of work for direct measurement of gross thrust-minus-drag for several jet configurations[41,42]. A critical bibliography of the literature on afterbody drag analysis is being prepared and it is expected that this may generate the need for further experimental work.

At RAE (B) (6 11.8) the planned investigation of rig-support effects in the measurement of afterbody drag at subsonic speeds is progressing. The rig and model are now being manufactured; a parallel afterbody will be tested first to provide basic information on the pressure distribution due to tunnel and rig interference in the RAE 3′ and ARA 9′ x 8′ tunnels.

The work at ONERA (Ch) (6 11.9) concerning afterbody testing with the supporting sting upstream of the model is further pursued including boundary layer control on the sting. (See also Reference 43.)

A new investigation is reported from BAC (6 11.11). Single and twin nozzle afterbody configurations have been tested[44] at supersonic Mach numbers over a wide range of jet pressure ratio. The results obtained have been partly analyzed. Further tests to measure forebody influence will be done.

Many related investigations on afterbody testing are referred to in Section 3 of this report.

Flow in junctions between bodies. The rather extensive work at ULICA (69.1) concerning three-dimensional effects in nominally "two-dimensional" flows is now completed and written up in thesis form.[45] See also Eurovisc Annual Report 1975, Job 8.5. The project from ARL (69.3) on viscous flow interaction studies at high Mach numbers is closed and will in the future be reported from AFFDL. The work undertaken so far is reported in many publications among which the most recent are References 46 and 47.

Unsteady flows. Techniques for measuring unsteady flows have already been discussed in Section 2. Here, some of the actual problems are briefly described, which might give an indication of what kinds of test are needed.

The work at NLR (6 12.1), which is a fundamental study on unsteady two-dimensional air loads in transonic flows, is continuing. Experimental exploration of the unsteady aerodynamic characteristics for supercritical airfoils is under way and, in cooperation with NASA Ames, an investigation is made of the effect of Reynolds number and tunnel walls on unsteady pressure measurements. Three new publications[48,49,50] are announced. At Volvo (6 12.2) the experimental investigation, performed in a watertunnel, of unsteady aerodynamic forces on two-dimensional wing with control surfaces has now successfully been carried out. Reasonably good agreement with theoretical results have been obtained. Further tests with lower frequencies and studying the influence of amplitude variation will be made as well as tests with fixed wing and oscillating flaps.

At ONERA (CERT) a fundamental study of unsteady turbulent boundary layers has been undertaken (6 12.4). Interesting results are reported on the structure of turbulence. Theoretical prediction methods using finite difference techniques and elaborate turbulence modelling or simple integral methods have been developed. A report[51] is published.

Data from the planned investigation at NYU (6 12.6) of unsteady laminar and turbulent boundary layers in a tube-type windtunnel at low speeds is being obtained. Tests with both laminar and turbulent boundary layers and some heat transfer measurements have been made.

Two new investigations are finally reported on unsteady flows. The first, at AFFDL (6 12.7), has the objective to provide experimental flutter data in order to establish the reliability of analytical methods. A wing-fuselage-tail flutter model will be tested in the AEDC 4T windtunnel in the low supersonic speed regime ($M = 1.3$). Thickness effects will be investigated. The final technical report is under way.

The second investigation at Science Application, La Jolla, (6 12.8) aims at extending a computer program[52] for the unsteady aerodynamic forces on two-dimensional wings in transonic flow to include slightly supersonic speeds and wind-tunnel wall interference. It will also be extended to three-dimensional wings. The work of programme is partly completed.

Conclusions and recommendations. In the area of "fluid motion problems" much work of importance for design and operation of large windtunnels is going on apart from that reported here. It has not been possible, however, to cover everything and furthermore much relevant information may be found in other sources such as the Eurovisc Annual Reports, the AGARD FDP and FMP Conferences and Symposia and the AIAA Conferences and similar meetings. Especially should be mentioned the AGARD FDP Symposium on Wind Tunnel Design and Testing Techniques in London, October 1975 (Ref.2) and the Technical Evaluation Report,[3] that followed later.

In the present report, with its about 50 contributions on fluid motion problems, the more fundamental work in this area is reasonably well covered and many new contributions of great interest have been added since the last review.[53]

The need for more research on the various mechanisms of transition must, however, again be stressed. This area is of central importance to the objectives of the Subcommittee on Wind Tunnel Testing Techniques (TES). The Recommendations for Work on Transition in Boundary Layers, Appendix 7, of the present report will help the planning of future research on this subject. The AGARD Symposium on Transition in Copenhagen in May 1977 can also be expected to present results of great interest. Related to the area is the problem of the flow equality in windtunnels and also here more research is needed before the question of admissible disturbance level is settled.

Another area, of at least equal importance for our objectives, is the comparison between results obtained in windtunnels and in flight. Many investigations, some of them new, are under way and should contribute to further understanding and progress in the subject. Considering the importance of this area there are, however, strong motives and needs for much more work. This should be encouraged by the TES Subcommittee. In this context the recommendation No.2 in the Technical Evaluation Report[3] from the AGARD London Symposium 1975 should also be considered. That recommendation is: "The Fluid Dynamics Panel should take the lead in developing aerodynamic programs, both experimental and theoretical, to gain more knowledge about sensitive flow regimes around aircraft, particularly in the transonic speed range. Adequate knowledge would allow to define those test conditions which can be correctly simulated only in special large windtunnels, or in free flight, and most importantly, make it possible to design and safely operate aircraft throughout their entire speed range."

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PART III

MAIN CONCLUSIONS AND RECOMMENDATIONS

The magnitude of resources devoted to research on windtunnel design and testing techniques in the NATO nations is barely adequate to meet the demands for increased data variety and precision. Fortunately, the members of TES are in a position, in their respective nations, to influence the research undertaken so that the knowledge which they gain through their work with TES assures low redundance and high effectiveness of the overall program in NATO. This is evidenced by content of the program discussed in this report.

While the investigations discussed in this report should result in substantial improvements to testing techniques, further developments of major benefit to aeronautics remain feasible and should be pursued as resources become available. Needs for such advances and the possibility for achieving the further technology gains have been developed and are indicated in the following:

(1) Development of test techniques, instrumentation, and analysis methods related to static and dynamic stability including cross coupling between longitudinal and lateral motions of aircraft and missiles at important high angle of attack flight conditions of modern aircraft and missiles. (FDP will hold a symposium on this subject May 1978.)

(2) Development of improved capability for modeling, instrumentation, and data interpretation in aero-acoustic investigations. (An "international workshop" of the experts in the field is recommended.)

(3) Continued study of materials, surface finishes, design methods, instrumentation, and data acquisition systems for models, engine simulators and support systems for new high Reynolds number wind tunnels including those operating at cryogenic temperatures. (An FDP Round Table Seminar on this subject is being considered for spring 1979.)

(4) Establishment of transonic windtunnel wall interference correction methods and alleviation methods as well as standardized models and test programs for use in wall interference investigations. (FDP members have commented on a standardized model and program study. TES is to submit a paper on the model and program for discussion of FDP.)

(5) Further study of the influence of flow quality on windtunnel experiments. (FDP Symposium on Windtunnel Design and Testing Techniques October 1975, CP-168.)

(6) Study of turbulent boundary layers and their initiation through instability and transition using theory, wind-tunnel data and flight data. (FDP Symposium on Laminar Turbulent Transition May 77 and TES support of AEDC-NASA flight test of instrumented cone.)

(7) Application of non-obtrusive instrumentation, including laser type, for measurements of flow field temperature, pressure, velocity, density, and composition in separated and other flow regions needed to yield new understanding of such flows. (Work is underway at NASA, AEDC, RAE, and ONERA and as that work yields results FDP discuss provision for dissemination of the results and evaluation of the impact of those results.)

(8) Further development of stall, departure, and skin test techniques using combined windtunnel and computer simulation of aircraft.

PART IV

PROGRAM OF WORK

Note: Asterisk denotes updated job cards have not been received.

1	**WINDTUNNEL DESIGN AND OPERATION**	
10	**Design of windtunnels**	
10.1	Assessment of design requirements of the test section of the proposed Large Subsonic Windtunnel (LST 8 x 6) of the NLR on the basis of experiments in the model tunnel.	NLR B.M.Spee F.Jaarsma
10.2	Construction and instrumentation of a pilot tunnel employing ECT-drive and investigation of its flow and performance characteristics.	RAE (B) P.G.Pugh
10.3	Provision of 5 m tunnel	RAE (F) A.Spence
10.4	Improvements to RAE 24 ft Tunnel	RAE (F) T.B.Owen
10.5	Constant pressure storage	FFA C.Nelander
10.6	Project Transonic Ludwieg Tube	DFVLR (G) W.Lorenz-Meyer
10.7	Work about design and operation of the new subsonic pressurized tunnel F1 – Le Fauga	ONERA (Ch) J.Christophe
10.8	Construction and instrumentation of pilot tunnels T_2 and T'_2 to demonstrate the Injector Driven	ONERA (Ch) J.P.Chevallier
10.9	Carrying on the Project Grosser Untersschall-Kanal GUK	DFVLR (PW) Pfeiffer
10.13	HIRT Advocacy Studies	AEDC C.J.Schueler
10.14	Evaluate the injector drive concept for possible use in a large high Reynolds number transonic windtunnel	ARC Lado Muhlstein, Jr
10.15	Investigation of the application of the cryogenic concept to high Reynolds number transonic windtunnel	LaRC R.A.Kilgore
10.16	Laser powered windtunnel	Wash U A.Hertzberg
10.18	Transonic aerodynamic testing utilizing sled test vehicles	6585 TESTG T.R.Bruce
10.22	Model studies of various exhaust deflector schemes for the NAE 5 x 5 ft blowdown windtunnel	NAE D.Brown
10.23	Cryogenic blow-down or induced-flow windtunnel concept	FFA C.Nelander
10.24	Modification to 40- by 80-foot windtunnel: repowering and addition of an 80 x 120 ft test section	ARC K.W.Mort
10.25	Studies of wind-angle diffusers	ULICA P.Bradshaw

10.26	Influence of model and sled drag on flow corridor convergence. Methods for reducing the rate of flow corridor convergence.	Stuttgart Theo Hottner
10.27	Screen replacement in the NAE 5 ft x 5 ft blowdown windtunnel.	NAE R.H.Piper
10.28	Mach number control system for subsonic and transonic operation of the NAE 5 ft x 5 ft blowdown windtunnel.	NAE L.H.Ohman
10.29	Construction of low speed windtunnel DNW	NLR/DFVLR F.Jaarma
12	**Review of current methods and development of new methods of constructing rigid and elastic models**	
12.1	Design study of representative high-lift aircraft model for RAE 5 metre tunnel	RAE (F) A.Spence
12.3	Design of a large low speed windtunnel model.	Saab T.Örnberg
12.4	Statically aeroelastic model of reinforced plastic	FFA S.Lundgren
12.6	Elastic windtunnel models of a supersonic fighter aircraft for static measurement in trisonic windtunnels.	Saab T.Örnberg
12.7	Dynamically correct models for high-speed investigations	Saab B.Åkerlindh
12.8	Study of glass and carbon reinforced plastic construction for model helicopter blades with dynamically scaled characteristics.	RAE (FS) A.Anscombe
12.9*	Improvement in the design and manufacture of models with representative mass distribution intended for free-flight tests.	IMF (L) J.Gobeltz F.Dupriez
12.10*	Contribution to the study of making a fuselage of the right stiffness for flutter models	IMF (L) J.Gobeltz F.Dupriez
12.11*	Contribution to the study of dynamic simulation in a flutter model of a partly-filled external fuel tank	IMF (L) J.Gobeltz F.Dupriez
12.12	Work about the design and building of models	ONERA (Ch) M.Bazin P.Broussaud
12.13	Design, construction and flutter test of aeroelastic similar models of large commercial airplanes	ONERA (Ch) R.Destuynder
12.14*	Modelling technique for static aeroelastic similarity for supersonic high-pressure blowdown windtunnel.	CRA C.Buongiorno U.Ponzi
12.15	Development of techniques for construction of test articles for simulated aerothermodynamic testing of weapon system concepts.	AFFDL E.L.White
12.16	Design, construction and test of models representing aeroelastic effects on steady forces and moments	RAE (F) G.F.Moss
13	**Review of new methods for supporting models, including the effect of rate of change of model attitude on measurements**	
13.2	Development of new supports for Modane windtunnels	ONERA (Ch) J.Christophe M.Bazin
13.3	Effect of rate of change of model attitude on force measurements	NAE E.Atraghji J.R.Digney

13.4	Aerodynamic carriage loads study	AEDC R.E.Dix
13.5	Support interference in transonic windtunnels	JPL B.Dayman
13.6	Store separation testing criteria	AEDC R.E.Dix
13.7	Prediction of model support interference effects	NLR C.Rip
13.8	Recording of the trajectory of a store released from a parent aircraft model by twin ejector guns. Variable are Mach number, aircraft attitude, simulated attitude and gun energy distribution. Light store modelling techniques used a high speed and . . .	BAC (Wa) E.S.Greening C.Russel
14	**Methods for data acquisition and analysis**	
14.1	Data acquisition and handling by a big central computer in real time.	DFVLR (G) D.Mehmel
14.2	Data assembling with computing systems and on line distribution of the results.	DFVLR (BF) F.W.Scholkemeier
14.3	Planning of a new system for acquisition and reduction of data from the transonic-supersonic windtunnel S4 at FFA	FFA K.Fristedt
14.4	5 m tunnel instrumentation	RAE (F) R.Jeffery
14.5	Unsteady data acquisition	RAE (F) R.J.North
14.6*	Acquisition and analysis of data transmitted by radiotelemeters and cables during free-flight model tests.	IMF (L) J.Gobeltz J.P.Druel
14.7*	Integration of a system for the handling of telemetered data in the flight loop of a model in a free-flight	IMF (L) J.Gobeltz J.P.Druel
14.8	Development of the technique of measurement and data analysis	ONERA (Ch) R.Tisseau ONERA (M) J.Fiquet
14.9	Acquisition and processing of unsteady data	NLR H.Tijdeman
14.10	Development of a generalized data reduction system	NAE R.D.Galway
14.11	Data acquisition system for NAE 5 x 5 ft windtunnel	NAE A.J.Bowker
14.12	Application of math models to windtunnel testing	AEDC R.L.Palko
15	**Investigation of unconventional design for low-speed windtunnels**	
15.1	Quasi-continuous low-speed tunnel operating at high pressure	FFA C.Nelander
15.3*	Study relating to the design of a large low-speed windtunnel for tests on catapulted free-flight models on aircraft response to horizontal gusts.	IMF (L) J.Gobeltz R.Vergrugge
16	**Investigations of techniques for managing turbulence in windtunnels**	
16.2*	Diffusion and decay of turbulence created by wire gauze screens placed across the flow in a transonic windtunnel.	IMF (L) G.Gontier A.Dyment

16.3	Experimental work on the reduction of noise in a transonic blow down windtunnel	ONERA (Ch) J.P.Chevallier
16.5	Investigation of flow distribution and flow unsteadiness and means for its control in the NAE 5 x 5 ft blowdown windtunnel	NAE D.Brown D.Peake
16.6	Measurements of flow quality in the transonic section of the NAE 5 x 5 ft windtunnel	NAE R.Galway D.Brown
17	**The design of anechoic working sections**	
17.1	Subsonic windtunnel design for model noise testing	RAE (F) J.Williams Holbeche
17.2	Development of a new facility for acoustic research at CEPr Saclay Center (CEPRA 19)	ONERA (Ch) J.Christophe
17.3	DNW subsonic windtunnel design for model noise testing	NLR J.C.A. van Ditshuizen
19	**The effect of contractions on boundary layer turbulence and thickness**	
1 10	**Acoustic resonance in large tubes and means for their suppression**	
1 11	**Scaling laws for wave motions in non-uniform ducts, including energy dissipation and heat transfer**	
1 12	**Investigation of real-gas effects when using air flows at sub-ambient temperatures**	
1 12.1	Heavy gas wind tunnel testing	ARC F.W.Steinle
1 12.3	Flow simulation for aerodynamic ground testing	AFFDL R.R.Smith
2	**GENERAL TESTING TECHNIQUES**	
21	**Techniques for measuring steady and unsteady pressures and forces**	
21.1	Investigations in the field of unsteady pressures with stochastic character	NLR H.Tijdeman
21.2	Measurements of steady pressure distributions in intake	DFVLR (BF) D.Christ
21.3	Provision for measurement of unsteady pressures	RAE (B) K.G.Moreton B.L.Welsh
21.4	Pressure measurements on harmonically oscillating wings, stabilizers, fuselages, and external loads	DFVLR (G) H.Triebstein
21.5	Measurements with oscillating two-dimesional aerofoils representing helicopter blades	RAE (B) N.C.Lambourne
21.6	Experimental work on dynamic distortion in air inlets	FFA K.A.Widing
21.7	Pressure measurements on air intake lips in low speed windtunnels	ITS S.O.Ridder
21.9	Design and operation of special transducers for unsteady pressure measurements	ONERA (Ch) E.Larguier
21.10	Electronic beam welded six-component strain gauge balances	Saab T.Örnberg

21.11	Measurement of steady and unsteady pressures on a wing model with pod-mounted engine	ONERA (Ch) R.Destuynder
21.12	Test proceedings for separation of pressure waves in turbulent flows	MBB W.Habig
21.13	Investigations of unsteady flow fields in supersonic intakes	MBB W.Habig
21.14	Development of aerodynamic force measurement techniques for testing models	AFFDL F.W.Little
21.15	Development of system for measuring pressure distribution on models	AFFDL E.B.Peters
21.16	The use of buoyant force balance tares	NAE R.D.Galway
21.17	Fast scanning pressure measurement techniques	NAE A.J.Bowker
21.18	Measuring pressure fluctuation levels in transonic windtunnels	DFVLR (G) W.Lorenz-Meyer F.R.Grosche
21.19	Measuring pressure fluctuation in transonic windtunnels	RAE (F) G.F.Moss
21.20	Moving probe for continuous measurements of air inlet flow distortions in low speed windtunnel tests	ITS S.O.Ridder
21.21	Development of a rig for measuring cowl drag via the wake momentum deficit	RAE (B) E.C.Carter
21.22	System for continuous indication of compressor face Mach number at air inlet tests	FFA K.A.Widing
21.23	Nonharmonic unsteady pressure measurements on rotating wings and propellers	DFVLR (G) K.Kienappel
21.24	Computer aided design for heavy duty multi-component balances	ONERA (Ch) M.Bazin
22	**Techniques for measuring and analyzing steady and unsteady flow fields**	
22.1	Investigations with the help of special techniques of pressure fields with a stochastic character	NLR van Nunen
22.2	Experimental and theoretical work on flow direction measurement with five-tube probes in gas flow	DFVLR (BB) R.Ulken
22.3	Measurement of steady flow fields by an automatic driven probe in a low speed windtunnel	DFVLR (G) H.J.Graefe
22.4	*Continuous traversing of temperature profiles*	DFVLR (BB) W.Alvermann
22.5	Smoke flow visualization at high wind speeds	DFVLR (G) W.Stahl
22.6*	Measurement of the transient downward deflection of a wing crossing the waves of vertical gusts	IMF (L) J.Gobeltz R.Verbrugge
22.7	Development of new optical methods for measuring boundary layers in shock tunnels and ballistic ranges. Analyses of pressure, heat transfer and structure turbulence.	ISL H.Oertel
22.8	Experimental study of unsteady boundary layers on an oscillating wing, using thin films and hot wires	ONERA (Ch) J.J.Philippe

22.9	Laser Anemometer (applied to subsonic and supersonic gaseous flows)	ISL B.Koch H.J.Pfeifer J.Haertig
22.10	Development of a pulsed wire and pulsed gauge anemometer for velocity and shear stress measurement in highly turbulent flows.	USME L.J.S.Bradbury
22.11	Experimental work on the application of laser Doppler to the study of laminar and highly turbulent flows in the range from 50 meters/sec to 3000 meters/sec.	UKP D.A.Jackson
22.12	Laser anemometry	RAE (F) J.B.Abbiss R.J.North
22.13	Laser-Doppler-Velocimeter (LDV) for trans- and supersonic flow	DFVLR (PW) F.Maurer
22.14	Measurement of surface shear with floating element transducers	UOES R.E.Franklin
22.15	Transonic flow visualization by bubbles	SAGE L.S.Iwan
22.17	Low Mach number aerothermodynamic investigations	AFFDL M.E.Hillsamer
22.18	Flow field probes for ground testing of high-speed aircraft and missiles	AFFDL F.J.Huber
22.19	Performance Evaluation of the AEDC probe in measuring local enthalpy in reentry test facilities	AFFDL W.E.Alexander
22.20	Laser anemometry	ULICME J.H.Whitelaw
22.21*	Upstream infinity	AMBDA J.C.Vayssaire
22.22*	Study of boundary-layer nature and of flow direction by means of hot films	ONERA (M) C.Armand
22.23	Measurements of local skin friction	FFA A.Bertelrud
22.24	Laser velocimeter	AEDC E.E.Newman
22.25	IR System for aerodynamic heating and transition measurements	AEDC D.S.Bynum
22.26	Raman-Rayleigh diagnostics	AEDC J.W.L.Lewis
22.27	Holographic interferometry	AEDC J.W.O'Hare
22.28	IR pyrometer for model surface temperatures	AFFDL E.L.White
22.29	Use of surface hot-films to detect transition, separation and location of shock-wave boundary-layer interaction	NLR R.Ross
22.30	Flow direction probe for measurements in low speed windtunnels	ITS S.O.Ridder
22.31	Laser anemometry (applied in windtunnels, free-jets, flames, compressors, etc. . . .)	ONERA (Ch) A.Boutier
22.32	Holographic interferometry	ONERA (Ch) J.Surget J.Delery

22.33	Raman and CARS diagnostics	ONERA (Ch) J.P.Taran
22.34	Non-obtrusive detection of transition region by infra-red camera	NAE D.J.Peake
23	**The measurement of static and dynamic derivatives**	
23.1	Measurements of control surface derivatives in transonic tunnels.	RAE (B) N.C.Lambourne
23.2	Development of methods for measuring dynamic derivatives in relatively small windtunnels with relatively dynamic pressure	DFVLR (PW) I.Niezgodka N.Treinies F.Maurer
23.3	Elaboration of test equipment for measuring dynamic stability derivatives on slender aircraft models in AVA-windtunnels	DFVLR (G) E.Schmidt
23.4	Development of a series of windtunnel balances and a calibration rig. Application of matrix methods on data reduction of calibration and test data	FFA K.Fristedt
23.5	Measurement of dynamic stability derivatives of Viking model in FFA windtunnel S4 at transonic and supersonic speeds	FFA S.Lundgren
23.6	Inertia-compensated balance for measuring transient aerodynamic disturbances from drop tanks.	Volvo R.Borg
23.7	Assessment of the long-term needs for dynamic testing and the development of suitable techniques	RAE (B) H.H.B.M.Thomas R.Fail
23.9*	Determination of static aerodynamic characteristics, using catapulted free-flight models	IMF (L) J.Gobeltz R.Verbrugge
23.10*	Determination of dynamic derivatives, using free-flight models and simulation on an analogue computer	IMF (L) J.Gobeltz R.Verbrugge
23.11	Technique of measuring unsteady aerodynamic derivatives by method of forced oscillations	ONERA (Ch) M.E.Erlich
23.12	Development and application of strain gauge balances	DFVLR (PW) A.Heyser P.J.Weber
23.13	Windtunnel measurements of dynamic stability derivatives on models of aircraft, missiles, and reentry bodies	ONERA (Ch) X.Vaucheret ONERA (M) M.Canu
23.14*	Effect of rotation in the spin on the aerodynamic coefficients	IMF (L) J.Gobeltz
23.15	To measure stability derivatives of a non-so-slender wing/fin configuration (see also item 23.5)	RAE (B) R.Fail
23.16	Aeromechanics, prediction and analysis	Fla U Leanon Clarkson Bullock
23.17	Aircraft/weapon performance, stability and control	AFATL R.E. Van Putte
23.18	Aircraft weapon separation analysis	AFATL C.B.Mathews
23.19	Reflection plane technique for dynamic stability testing	AEDC H.C.Dubose

23.20	Development of techniques for measuring dynamic stability derivatives on models at high angles of attack	NAE K.J.Orlik-Rückemann
23.21	Design and development of continuous rolling rigs for the measurement of dynamic derivatives due to rolling in high and low speed tunnels	BAC (Wa) P.G.Knott RAE (B) R.Fail
24	**The measurement of aerodynamic and structural damping and of the frequency response to disturbances**	
24.1	Investigations of methods to obtain correct values of damping and frequencies	NLR H.Tijdeman
24.2	Elaboration of a measuring procedure and equipment for flutter investigations in windtunnels	DFVLR (G) P.Bublitz
25	**Technique for measuring aeroelastic and flutter characteristics**	
25.1	Evaluation of methods for excitation of windtunnel models	NLR H.Tijdeman
25.2	Comparison between theoretical and experimental flutter speed of T-tails in the high speed region, where the influence of angle of attack is studied	Saab V.J.Stark B.Åkerlindh
25.3	Ground vibration tests of dynamical models	Volvo R.Frankmark
25.4	Measurement of flutter aerodynamics for a wing of modern design	RAE (B) N.C.Lambourne ONERA R.Destuynder
25.5	Development of methods for use in short-duration facilities	RAE (FS) C.Skingle D.Drane
25.6	Measurement of flutter characteristics of a wing with pod-mounted engine. Comparison with theory	ONERA (Ch) R.Destuynder
25.7	Direct model attitude sensor	AEDC R.L.Ledford
25.8	Direct model attitude and shape sensor	AEDC W.H.Goethert
25.9	Optical arrangements for the movement of turbo-machine blade vibrations	ONERA (Ch) M.Philbert
26	**Techniques for simulating and measuring transient motions, such as gusts**	
26.1*	Experiments to determine the response of an aircraft to atmospheric gusts, using free-flight models	IMF (L) J.Gobeltz R.Vergrugge
26.4	Gust simulation in a windtunnel	ONERA (Ch) J.Christophe
26.5	Simulation of gusts in windtunnels	RAE (B) J.G.Jones
26.6	Development of a gust generator	DFVLR (BF) D.Christ
27	**Techniques for measuring ground effects**	
27.1	Investigation on effects associated with the representation of the ground by a fixed board in windtunnel tests	NLR S.O.T.H.Han
27.2	Ground effects on windtunnel measurements	DFVLR (BF) Schroeder

27.3	Recirculation flow of VTOL lift engine	DFVLR (BB) E.Schwantes
27.4*	Free-flight model experiments to determine the nature of ground effects in calm air, including effects during descent	IMF (L) J.Gobeltz R.Vergrugge
27.5*	Free-flight model tests on aircraft landing through a steady crosswind	IMF (L) J.Gobeltz R.Vergrugge
27.6*	Experimental means for studying, on catapulted free-flight models, the response of an aircraft to lateral gusts in ground effect at the end of approach	IMF (L) J.Bogeltz R.Verbrugge
27.8	Ground simulation in S1 Modane windtunnel with a blown ground board	ONERA (Ch) Ph.Poisson-Quint J.Christophe
27.9	Measuring ground effects by pressure distribution measurements on the ground itself	DFVLR (PW) G.Schulz G.Viehweger
27.10	Determination of testing limits for the measurement of ground effect on a two-D wing with slotted flap using a fixed ground plane	VKI J.Sandford
27.11	Investigation on a plate with uniform boundary layer suction for ground effects	DFVLR (G) R.Wulf
28	**Methods for determining spinning characteristics**	
28.1*	Research on a spin recovery criterion, using the application of moments, created by rockets in a vertical windtunnel	IMF (L) J.Gobeltz L.Beaurain
28.2*	Effect of rotation in the spin on the aerodynamic coefficients	IMF (L) J.Gobeltz
28.3*	Model and full-scale comparison of spinning results	IMF (L) J.Gobeltz L.Beaurain
28.5	Anlytical computation of spinning motion	VKI F.Haus
29.	**The design of rigs for testing rotary wings**	
29.1	Test rig for helicopter rotors at S2 Chalais Meudon (ONERA) aimed at tests at high advance ratios	ONERA (Ch) J.J.Philippe
29.2	Experimental study of helicopter rotors in a windtunnel	ONERA (M) C.Armand
29.3	The design of rigs for testing rotary wings	RAE (FS) A.Anscombe
29.4*	Visualization by threads on helicopter rotor blades in a windtunnel	ONERA (M) C.Armand
2 10	**Methods for measuring noise**	
2 10.1	Noise measurement techniques in windtunnels	RAE (F) Holbeche Williams
2 10.3	Methods for measuring noise	VKI J Sandford
2 10.4	Development of laser beam technique to measure turbulence in a jet	ISL H.J.Pfeifer

2 10.5	Acoustic measurements within a flow	DFVLR (PW) G.Schulz
2 10.6	Measurement of helicopter rotor noise in a windtunnel	ONERA (M) C.Armand
2 10.7	Jet noise suppressor testing in the small anechoic windtunnel of KAT of NLR	NLR W.B. de Wolf
2 10.8	Airframe aerodynamic noise	ARC D.H.Hickey
2 10.9	Space-time structures of acoustic fields	ONERA (Ch) M.Perulli
2.11	**Development of noise generators**	
2 11.1	Development and assessment of model noise generators	RAE (F) Holbeche Williams
2 11.2	Jet noise generation for windtunnel models	NLR W.B. de Wolf
2 12	**Techniques for simulating adverse weather conditions such as icing, rain erosion, etc.**	
2 12.1	Icing testing at fullscale and reduced scale in S1 Modane windtunnel	ONERA (Ch) G.Leclere ONERA (M) F.Charpin
2 12.2	Rain erosion testing in S3 Modane windtunnel	ONERA (Ch) G.Leclere
3	**SPECIAL TECHNIQUES FOR ENGINE SIMULATION**	
30	**Comparison of techniques for engine simulation**	
30.1	Survey of ARA experience of flow simulation for underwing nacelles	ARA E.C.Carter
30.2	Problems and test techniques associated with integrated nozzle-afterbody testing in transonic windtunnels	AEDC L.L.Galligher
30.3	Engine inlet testing at high maneuver conditions	AEDC R.L.Palko
30.4	Turbine engine/airframe exhaust system interaction	MDC (M) R.Martens
30.6	Multi-mission turbine-engine propulsion simulator application	AFAPL S.J.Piller
30.7	(1) Tests on a jet lift model over the transition flight regime to investigate the validity of momentum ratio scaling (contract funded) (2) Tests on a target type thrust reverser model in the landing touch down and ground roll regime to investigate the . . .	BAC (W) B.Earnshaw (RAE)
31	**Investigations into the possibility of operating engine simulators under pressurized conditions**	
31.1	Design and tests of engine simulators duplicate various by-pass ratios in pressurized tunnels	ONERA (Ch) P.Broussaud J.Christophe
32	**Development of methods to obtain quick engine/jet data**	
32.1	Data acquisition and computation system for model engine testing	DFVLR (G) E.Melzer R.Wulf

32.2	Flow investigation of hot engine jets	DFVLR (BF) A.Zacharias
32.3	Simulation of rocket-engine jets in relatively small windtunnel models	DFVLR (PW) F.Maurer L.France
32.4	Techniques of sonic and supersonic jet simulation by compressed air for models in low speed windtunnels	DFVLR (PW) Viehweger
32.5	Development of electronic systems for recording and simultaneous processing of instantaneous pressure distributions	MBB W.Habig
33	**Development of high-precision balances, with ducts for drive medium**	
33.2	Further development of techniques for afterbody testing	Volvo G.Rosander
33.3	Further development of a pneumatic balance for accurate measurement of drag with simulated inlet flow	Volvo G.Rosander
33.4	The development of an internal strain gauge balance in the presence of a compressed air supply	BAC (Wa) P.G.Knott
33.5	Design and operation of special balance with ducts for drive medium in Modane	ONERA (Ch) P.Broussaud
34	**Development of ejectors and powered nacelles driven by decomposition products of H_2O_2**	
34.2	Calibration tests in BAC Weybridge 13 x 9 low speed tunnel of ejector-driven RB211 simulator	RAE (F) J.A.Bagley BAC (Wey) D.J.Stewart
34.3	Tests of single and multiple-nozzle ejectors as basis for design of ejector-driven engine simulators	RAE (F) J.Crane BAC (Wa) P.G.Knott
34.4	Development of a high speed windtunnel technique for combat aircraft models having simultaneous engine intake and exit flow simulation	BAC (Wa) P.G.Knott RAE (F) J.Bagley
35	**Development of small highly-loaded compressors and turbines for integrated propulsion schemes**	
35.1	Calculation and construction of new fans for models in a 3 m low speed windtunnel	DFVLR (G) E.Melzer R.Wulf
36	**Development of advanced model fans with similar aerodynamic characteristics as full-scale engines and investigation of scaling laws and extrapolations**	
36.1	Design and tests of an engine simulator (by pass ratio 10)	ONERA (Ch) P.Broussaud J.Christophe
37	**Development of small real engines, with front area thrust similar to full-scale engines and having similar inner aerodynamics for noise measurements**	
38	**Development of systems for thrust vectoring and thrust reversing**	
38.1	Model investigations on jet cascades for thrust vectoring	DFVLR (G) E.Melzer R.Wulf
38.2	Windtunnel tests for external aerodynamic studies of thrust reversal at ground roll	Saab G.Hellström

39	**Techniques for adding heat to airflows in windtunnels**	
39.1	Development of a small propane-oxygen gas generator to simulate hot and fuel-rich primary jets	DFVLR (BB) E.Riester
39.3	Hot versus cold plume simulation for jet engines	AEDC C.E.Robinson
39.4	Hot versus cold plume simulation for military jet engines	AFFDL P.C.Everling
4	**SPECIAL TECHNIQUES FOR HIGH-LIFT AND V/STOL TESTING AT LOW SPEEDS**	
41	**Investigation of the test limitations due to flow breakdown and of the effects of flow breakdown on model characteristics, in test sections with closed and with porous walls**	
41.2	Windtunnel constraint on models with lift jets	RAE (F) T.B.Owen
42	**Wall corrections and limits of applicability, including models for the curvature of jets discharging across the stream, for different types and sizes of test section and measurements of vorticity in the wake**	
42.1	Application of windtunnel corrections to measurements in open and closed test sections	DFVLR (BF) H.Otto
42.2	Windtunnel constraint on models with lift jets	RAE (F) T.B.Owen
42.4	Displacement corrections for large models with extended wakes	DFVLR (PW) G.Schulz
42.5	The study of operational problems and techniques in windtunnel testing of VTOL and STOL vehicles 10655	Wash U W.H.Rae
42.6	The application of vortex lattice techniques to the estimation of boundary corrections for a high-lift wing mounted in a windtunnel with an open test section	VKI H.Wirz J.Sandford
42.7	Computation of the wall-induced upwash-distribution for swept wings in a circular low speed tunnel	FFA S.Hedman
42.8	Effects of wake blockage when testing high-lift models in low speed windtunnels	ITS S.O.Ridder
43	**The design of slotted or porous test sections, including investigation of the possibilities of designing test sections yielding zero lift and pitching-moment corrections for V/STOL configurations, and of the effects of finite slot length and slot shape and of the sensitivity of the designs to changes of model configuration**	
43.1	Investigation of test section configurations for the large scale V/STOL windtunnel	AEDC F.L.Heltsley
43.2*	Theoretical lift interference study using vortex lattice method	AMBDA J.C.Vayssaire
43.3	Theory and experiments for new slotted-wall configurations for 2D aerofoil testing	UBCM G.V.Parkinson
43.4	Theoretical and experimental investigations of slotted wall test sections	NLR R.A.Maarsingh
44	**Extension of the methods for measuring ground effects to V/STOL models in the range of heights where a moving belt is not required because the boundary layer on a fixed plate does not separate**	

45	**Study of the problem of recirculation of disturbances in the tunnel circuit, including investigations of the effects of large wakes, such as those produced by V/STOL types of models on diffuser performance, as well as the effects of screens, honeycombs, fans and changes in the cross-sectional area on such disturbances**	
46	**Techniques for two-dimensional and half-model testing**	
46.1	Two-dimensional measurements on wing sections with high-lift devices	DFVLR (BF) Amtsberg Schroeder
46.2	2-dimensional testing of high-lift devices in the 12 ft low speed windtunnel	FFA Ingelman-Sundberg
46.3	Half-model high-lift techniques for the FFA 12 ft diameter low-speed tunnel	FFA Ingelman-Sundberg
46.4	Use of blowing slot to avoid parasitic separation on the walls of a two-dimensional test section	ONERA (Ch) B.Monnerie
5	**SPECIAL PROBLEMS OF TESTING AT TRANSONIC SPEEDS**	
51	**Experimental checks of analytical corrections for tunnel-wall interference**	
51.2	Experimental determination of wall interference in a transonic profile test section	DFVLR (BF) W.Puffert
51.3	Methods based on modulation principle	VKI Smolderen
51.5	Detailed pressure distribution measurements on NACA 0012 profile in different two-dimensional transonic windtunnels	ONERA (Ch) J.P.Chevallier
51.6*	Check of applying wall corrections to two-dimensional flow	IMF (L) Gontier A.Dyment
51.7	Tests on aerofoils in two-dimensional transonic flow	ONERA (M) X.Vaucheret (Ch) P.Marion (M)
51.8	Wall interference effects in two-dimensional flow	NAE M.Mokry
51.9	Determination, for given wall specifications of the porosity parameter	ONERA (Ch) J.P.Chevallier
51.10	Experimental investigation of slotted liner performance at high-subsonic speeds	RAE (B) M.N.Wood
51.11	Tunnel and scale effects on transonic flow wall interference	ARC F.W.Steinle
51.12	Investigation of wind-tunnel wall-interference effects near $M = 1.0$	LaRC W.B.Compton, III
51.13	Transonic windtunnel wall interference and advanced airfoil concepts	GD H.Yoshihara
51.14	Fundamental investigations in viscous transonic flow	JPL D.J.Collins
51.15	Reflection-plane model testing technique in transonic windtunnels	FFA S.E.Gudmundson
52	**Investigation of the effect of streamline curvature, with a view to determining the curvature coefficient of pitching-moment corrections at any point in the test section**	

52.1*	Analytical lift interference, including pitching-moment correction	AMBA J.C.Vayssaire
53	**Determination of the limits of applications of present correction methods, including boundaries for α and M for given model size (in combination with experiments on ONERA calibration models, see also 5 13)**	
53.3	Windtunnel wall corrections for three-dimensional models in transonic flow	ONERA (Ch) X.Vaucheret
53.4*	Sonic blockage/choked windtunnel	AMBDA J.C.Vayssaire
54	**Development of calculation methods to determine the influence of the finite length of the test section and of model position on the wall interference, including spanwise variations**	
54.1	Theoretical wall interference study	NLR J.Smith
55	**Collection of porosity data in data sheets, giving the porous wall specifications, with the objective to select porosity parameter schedules and to prove their value for many different models**	
56	**Research to determine the influence of the boundary-layer thickness, relative to hole diameter, on the porosity factor, including the influence of the stagnation pressure and of auxiliary plenum-chamber suction**	
56.2	Advanced transonic test section walls	ARC F.Steinle
56.3	Wall interference effects in transonic flow	CAL R.J.Vidal
57	**Determination of an ideal porosity schedule for zero interference or shock cancellation as a function of Mach number, including the effect of deviations from the ideal schedule**	
57.2*	Study of the shock development in two-dimensional flow for different Mach numbers and different types of wall perforation	IMF (L) G.Gontier A.Dyment
57.3*	Reflection and absorption of shock waves	AMDBA J.C.Vayssaire
58	**Investigation of wall porosity behaviour at supercritical speeds when shock waves are present, including the effects of shock waves impinging on the boundary layer along the wall on porosity behaviour**	
59	**Development of a theory for determination wall corrections at Mach numbers near unity**	
59.1	Transonic flow at a slotted test section wall	FFA S.Berndt
59.2	Determination of wall corrections for Mach numbers close to unity by using the numerical solution of the transonic small-perturbation equation	ONERA (Ch) J.P.Chevallier
59.3	Interference due to perforated walls at transonic speeds	Oceanics T.R.Goodman
59.4	Computations of wall effects in transonic windtunnels	RAE (F) D.Catherall
59.5	Analysis of transonic flow	Ariz U W.R.Sears
5 10	**Determination of wall interference corrections by application of numerical methods, with particular reference to viscous effects and three dimensional flows**	

5 10.1	Use of time dependent principle	VKI Smolderen
5 10.2	Theoretical wall interference study	NLR J.Smith
5 10.3	Establishment of capability for inviscid transonic flow field computation at AEDC	AEDC J.L.Jacocks
5 10.4	Effects of tunnel wall porosity at supercritical Mach numbers	AEDC J.L.Jacocks
5 10.6	Windtunnel wall interference under high lift conditions appropriate to transonic maneuvering of air superiority aircraft	AFFDL A.W.Fiore
5 10.7*	Theoretical lift interference study using vortex-lattice method	AMDBA J.C.Vayssaire
5 10.8	Wall-interference assessment technology for transonic windtunnels	LaRC W.B.Kemp, Jr
5 11	**Research into variable-geometry walls, especially streamwise distributions of porosity, with a view to obtaining zero interference for lift and moment at the same time, including the selection of an appropriate choice of ventilated walls as it depends on the interference that should be minimized and on the type of model**	
5 11.2	Development of an interference-free, self-adjusting, transonic windtunnel	CAL R.J.Vidal
5 11.3	Wall interference in transonic windtunnels	ATL P.Baronti
5 11.4	Theoretical and experimental study of self-correcting geometric local deflection and suction distribution to satisfy unlimited flow conditions	ONERA (Ch) J.P.Chevallier
5 11.5	Self streamlining windtunnel	USAA M.J.Goodyer
5 11.6	Wave attenuation for interference-free testing at transonic speeds	USAA M.J.Goodyer
5 11.7	Adaptive transonic wall study	AEDC E.M.Kraft
5 11.8	(a) Shock wave cancellation at elastic walls (b) Investigation of instationary shock wave interaction with a subsonic boundary	Stuttgart T.Hottner
5 11.9	Design study of a convertible transonic test section	FFA S.E.Nyberg
5 12	**Investigation of the noise generated by ventilated walls, including the clarification of the various edge-tone cavity-response mechanisms, which generate flow disturbances, the possible means for reducing flow disturbances and the influence of disturbance on model results**	
5 12.1	Studies of disturbances caused by perforated liners at transonic speeds	ULC Freestone
5 12.2	Measurement of the intensity level and spectral analysis of the noise in windtunnels	ONERA (Ch) J.P.Chevallier X.Vaucheret
5 12.4	Environmental noise in transonic windtunnels	Nielson J.P.Woolley
5 12.5	Perforated wall acoustic parameter study	AEDC N.S.Dougherty

5 13	**Experimental determination of wall interference by means of testing a series of standard calibration models in different European and American facilities, as proposed by ONERA**	
5 13.2	Three-dimensional slotted-wall interference investigation in the HST	NLR J.Smith
5 13.3	Comparative tests with ONERA calibration models in three FFA windtunnels	FFA S.E.Gudmundson
5 13.5	Comparison of European and American transonic windtunnels by means of similar standard models	ONERA (Ch) X.Vaucheret
5 13.6	Experimental wall interference study of NAE 5 ft x 5 ft windtunnel	NAE R.D.Galway M.Mokry
5 13.7	Investigation of windtunnel blockage and support interference effects for winged-body models	LeRC D.Bowditch
5 13.8	Missile aerodynamic methods at transonic speeds	MARTIN J.Fidler
5.14	**The design of the plenum chamber**	
5 15	**The effect of heat transfer to models in test conditions**	
5 15.1	Effects of heat-transfer on the characteristics of aerofoils at subsonic speeds	RAE (B) J.E.Green
6	**FLUID MOTION PROBLEMS**	
61	**The mechanism of transition**	
61.1	*Theoretical and experimental work on transition in three-dimensional* boundary layers	DFVLR (PW) E.H.Hirschel
61.2	Measurements of boundary layer transition on a cone, and of flow disturbances in the test section of the HST	NLR R.Ross
61.3	Transition detection and correlation	RAE (B) D.G.Mabey
61.5	Skin friction measurements on a two-dimensional aerofoil	NAE D.Brown
61.6	Experimental investigation of the effects of transition trips	FFA A.Bertelrud E.J.Totland
62	**The effects of velocity, pressure and temperature fluctuations and acoustically-excited disturbances on laminar and turbulent three-dimensional boundary layers**	
62.1	Experimental investigation into the effect of acoustical disturbances on separation	NLR R.Ross
62.2	The response of a turbulent boundary layer to acoustic excitation	RAE (B) D.J.Weeks
62.3	Windtunnel tests on a slender cone	AFFDL A.J.Murn
62.4	Correlation of transition Reynolds number with noise and turbulence levels in transonic tunnels	AEDC N.S.Dougherty
62.5	Turbulent boundary layer study	AEDC J.A.Benek
63	**The effects of surface imperfections on boundary layers**	

63.1	Excrescence drag	RAE (B) L.Gaudet
63.3	Windtunnel model surface effects study	UTSI J.Wu
64	**Techniques for simulating flows at higher Reynolds numbers**	
64.1	Technique for the use of transition stripe in high-lift testing	FFA Ingleman-Sundberg A.Bertelrud
64.3	Studies of scale effects on transonic flows on swept wings	RAE (B) C.R.Taylor
64.4	Exploratory investigation into the effects of compressibility at high-lift, low-speed	RAE (F) D.A.Kirby
64.5	Development of a technique for a detailed study of the leading edge (method of enlarged leading edge), and of the flap of a wing section	ONERA (Ch) B.Monnerie
64.6	Reynolds number effect experiments	LeRC D.Bowditch
64.7	High Reynolds number transonic testing techniques	OSU J.D.Lee
64.9	Scale effects on swept wings	ARC F.W.Steinle
64.12	Windtunnel measurement of the influence of roughness on an aircraft model with a thick aerofoil in transonic flow	ONERA (Ch) X.Vaucheret
65	**Conditions for separation in three dimensional flows, including shock-induced separations**	
65.2	Experiments on normal shockwave/boundary layer interactions	NLR J.W.Kooi
65.5	Transonic scaling of shock-boundary layer interaction	AEDC M.C.Altstatt
65.6	High Reynolds number tests of a NACA 65_1-213 and NASA 10-percent-thick supercritical airfoils at transonic speeds	CAL R.J.McGhee
65.7	Theoretical studies of shock-boundary layer interactions and boundary layer separation	VPI G.R.Inger
65.8	Transonic separated flow studies	CAL R.J.Vidal
65.9	Reynolds number effects and influence of upstream disturbances on the boundary-layer separation in transonic flow	UTSI J.M.Wu
66	**The consequences of separation in three-dimensional flows**	
66.1	Study of the prediction of buffet characteristics	NLR S.O.T.H.Han
66.2	Studies of various cases of three-dimensional flow separation	NAE D.J.Peake
66.3	Drag due to regions of compressible turbulent separation	BOEING H.Mansop
67	**The relaxation of turbulent boundary layers downstream of reattachment**	
67.1	Relaxation of turbulent boundary layers downstream of reattachment	ULICA P.Bradshaw
67.3	Experimental and theoretical work on shock-induced boundary-layer separation, reattachment and subsequent trailing-edge separation at transonic speeds	DFVLR (G) E.Stanewsky

68	**The flow in the near wake**	
68.2	Effect of three-dimensional disturbances on the spatial instability of compressible two-dimensional wakes	NAE Y.Y.Chan
69	**Three-dimensional flows in junctions between bodies**	
69.1	Three-dimensional effects in "two-dimensional" flows	ULICA P.Bradshaw
69.2	Flow in streamwise corners and wing-body junctions	ULQMC A.D.Young
69.3	Viscous flow interaction studies at high Mach numbers	ARL R.H.Korkegi R.Newman
6 11	**The effect of jets on neighbouring surfaces, including afterbody problems**	
6 11.1	Effect of some jet parameters on thrust minus drag of an axisymmetric afterbody with convergent nozzle at transonic speeds	NLR F.Jaarsma
6 11.2	Development and experimental evaluation of a method for the calculation of the aerodynamic interference between jets and aircraft parts outside the jets	NLR
6 11.3	Experiments on the determination of the lift distribution and the wake flow field of a wing immersed in the jets from propulsion systems	NLR R.A.Maarsingh
6 11.4	Development of a method for the calculation of the pressure distribution on an aerofoil in a two-dimensional non-uniform shear flow	NLR R.A.Maarsingh
6 11.5	Afterbody and near wake studies for vehicles with central propulsive jet; freestream interactions, including flow separation	FFA R.A.White S.E.Nyberg
6 11.6	Research on afterbody drag at transonic and supersonic speeds	RAE (F) J.Reid
6 11.7	Afterbody drag at transonic and supersonic speeds	ARA E.Carter
6 11.8	Investigation of rig support effects in the measurement of afterbody drag at subsonic and supersonic speeds	RAE (B) E.L.Goldsmith
6 11.9	Tests on rear bodies. Installation of an upstream support; control of the boundary layer on the support	ONERA (Ch) J.Leynaert
6 11.10	Design criteria of sub-scale windtunnel tests of jet interaction control effectiveness in flight	MDC (C) L.A.Cassel
6 11.11	Development of a rig to measure nozzle thrust and afterbody drag in the presence of jet afflux and external flow round representative fuselage shapes	BAC (Wa) A.Watson
6 12	**Unsteady inviscid and viscous flows**	
6 12.1	Unsteady airloads in transonic flows	NLR H.Tijdeman
6 12.2	Experimental determination of non-stationary aerodynamic forces on a two-dimensional wing with control surface in an incompressible fluid; a water tunnel is used	Volvo R.Frankmark
6 12.3	Study of the buffet of two-dimensional stalled aerofoils	RAE (B) L.F.East
6 12.4	Theoretical study of unsteady boundary layers on oscillating wing-sections	ONERA (Ch) J.J.Philippe
6 12.5	Experimental investigation of transonic buffeting of supercritical jet-flapped aerofoils	NAE D.J.Peake
6 12.6	Unsteady viscous flows	NYU J.G.Hall

6 12.7	Investigation of supersonic wing-tail flutter	AFFDL L.J.Huttsell
6 12.8	Separate steady and unsteady transonic flow solutions	SA R.M.Traci
6 13	**Comparison between results obtained in the laboratory and in flight**	
6 13.1	Thrust and jet flow measurements under laboratory and flight conditions	DFVLR (BF) Roscher
6 13.2	"Gnat" tunnel/flight comparison	RAE (B) D.G.Mabey J.G.Jones
6 13.3	Correlation between windtunnel results, theoretical and simulator predictions, and flight tests	ONERA (Ch) Ph.Poisson-Quinton J.Lerat B.Monnerie
6 13.4	Wing buffet onset for NACA 65_1-213 $(a = 0.5)$ section determined from windtunnel tests at flight values of Mach number and Reynolds number	NAE D.Brown
6 13.5	Inlet performance testing criteria	AEDC J.L.Jacocks
6 13.6	Effects of Reynolds number on installed performance of exhaust nozzle in windtunnels and flight	LeRC D.N.Bowditch
6 13.7	High Reynolds number testing (Lockheed Georgia CFF and NASA-MSFC Ludwieg)	ARC F.W.Steinle
6 13.8	Aerodynamic flight research analysis and data correlation	AFFDL W.A.Baldwin
6 14	**Investigation of boundary layers at very high Reynolds numbers**	
6 14.1	Experimental techniques associated with the determination of turbulent skin friction	NAE V.Ozaragpolu N. duy Vinh J.Dickinson
6 14.2	Experimental investigation of attached and separated flat plate turbulent boundary layers over a large Reynolds number range at subsonic speeds	NAE G.M.Elfstrom

LIST OF ABBREVIATIONS OF RESEARCH CENTERS

AECMA	Association Européenne des Constructeurs de Materiel Aérospatial, 88 Boulevards Malherbes, 75008 Paris, France
AEDC	Arnold Engineering Development Center, Arnold Air Force Station, Tennessee 37389, USA
AFAPL	Air Force Aero Propulsion Laboratory, Wright-Patterson Air Force Base, Ohio 45433, USA
AFATL	Air Force Armament Laboratory, Eglin Air Force Base, Florida 32542, USA
AFFDL	Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio 45433, USA
AMDBA	Avions Marcel Dassault, Bréguet-Aviation, 78 Quai Carnot, 92214 Saint-Cloud, France
ARA	Aircraft Research Association Limited, Manton Lane, Bedford, England
ARC	Ames Research Center, National Aeronautics and Space Administration, Moffett Field, California 84025, USA
Ariz U	University of Arizona, Tucson, Arizona 85721, USA
ARL	Aerospace Research Laboratory, Wright-Patterson Air Force Base, Ohio 45433, USA
ATL	Advanced Technology Labs Inc, 400 Jericho Turnpike, Jericho, New York 11753, USA
BAC(Wa)	British Aircraft Corporation Limited, Warton, Lancashire, England
BAC(Wey)	British Aircraft Corporation Limited, Brooklands Road, Weybridge, Surrey, England
BOEING	The Boeing Company, Box 3707, Seattle, Washington 98124, USA
CAL	Calspan Corporation, PO Box 235, Buffalo, New York 14221, USA
CRA	CRA, Via Salaria 851, Roma, Italy
DFVLR(BB)	Deutsch Forschungs-und Versuchsanstalt für Luft-und Raumfahrt, Institut für Antriebssysteme, 33 Braunschweig, Bienroder Weg 53, West Germany
DFVLR(BF)	Deutsche Forschungs-und Versuchsanstalt für Luft-und Raumfahrt, Zentralabteilung Niedergeschwindigkeits-Windkanäle Abt. B, 33 Braunschweig, Flughafen, West Germany
DFVLR(G)	Deutsche Forschungs-und Versuchsanstalt für Luft-und Raumfahrt, Aerodynamiche Versuchsanstalt Göttingen, 34 Göttingen, Bunsenstrasse 10, West Germany
DFVLR(PW)	Deutsche Forschungs-und Versuchsanstalt für Luft-und Raumfahrt, 505 Porz Wahn, Linder Höhe, West Germany
DFVLR(T)	Deutsche Forschungs-und Versuchsanstalt für Luft-und Raumfahrt, Institut für Antriebyssysteme, Aussenstelle Trauen, 3105 Fassberg, Postfach, West Germany
Dornier	Dornier-GmbH, 799 Friedrichshafen, Postfach 317, West Germany
FFA	The Aeronautical Research Institute of Sweden, Ranhammarsvägen 12–14, PO Box 11021, S-16111 Bromma 11, Sweden
Fla U	University of Florida, Gainesville, Florida 32601, USA
GD	General Dynamics Corporation, Convair Aerospace Division, Box 80877, San Diego, California 92138, USA
IASC	Institut Aérotechnique de Saint-Cyr, 15 rue Marat, 78210 St-Cry-L'Ecole, France
IMF(L)	Institut de Mécanique des Fluides de Lille, 5 Boulevard Paul Painlevé, 59000, Lille, France
ISL	Institut Franco-Allemand de Recherches de Saint-Louis, BP No.301, 68 Saint Louis, France
ITS	Royal Institute of Technology, Fack, S-100 44 Stockholm 70, Sweden
JPL	Jet Propulsion Laboratory, 4800 Oak Grove Drive, Pasadena, California 91103, USA
LaRC	Langley Research Center, National Aeronautics and Space Administration, Hampton, Virginia 23885, USA

LeRC	Lewis Research Center, National Aeronautics and Space Administration, 21000 Brookpark Road, Cleveland, Ohio 44135, USA
MARTIN	Martin-Marietta Corporation, Orlando Division, Box 5837, Orlando, Florida 32805, USA
MBB	Messerschmitt-Bölkow-Blohm, Ottobrunn, 8 Munchen 8, West Germany
MDC(C)	McDonnel Douglas Astronautics Company, Western Division, 5301 Bolsa Avenue, Huntington Beach, California 92647, USA
MDC(M)	McDonnel Douglas Corporation, Box 516, Saint Louis, Missouri 63166, USA
NAE	National Research Council, NAE, Montreal Road, Ottawa K1A OR6, Canada
NIELSON	Nielson Engineering and Research Center, 850 Maude Avenue, Mountain View, California 94040, USA
NLR	National Aerospace Laboratory, National Lucht-en-Ruimtevaartlaboratorium, Anthony Fokkerweg 2, Amsterdam 1017, The Netherlands
NYU	State University of New York, Buffalo, New York 14212, USA
OCEANICS	Oceanics Inc. Technical Industrial Park, Plainview, New York 11803, USA
ONERA(Ca)	Office National d'Etudes et de Recherches Aérospatiales, 16 rue de Maupassant, 06400 Cannes, France
ONERA(Ch)	Office National d'Etudes et de Recherches Aérospatiales, 29 Avenue de la Division Leclerc, 92320 Châtillon, France
ONERA(M)	Office National d'Etudes et de Recherches Aérospatiales, Centre de Modane-Avrieux, 73500 Modane, France
OSU	Ohio State University, Research Foundation, Columbus, Ohio 54130, USA
RAE(B)	Aerodynamics Department, Royal Aircraft Establishment, Bedford, England
RAE(F)	Aerodynamics Department, Royal Aircraft Establishment, Farnborough, Hampshire, England
RAE(FS)	Structures Department, Royal Aircraft Establishment, Farnborough, Hampshire, England
SA	Science Applications, Inc, La Jolla, California 92037, USA
Saab	Saab Scania AB, S-581 88 Linköping, Sweden
SAGE	Sage Action Inc, PO Box 416, Ithaca, New York 14850, USA
SB & C	Société Bertin et Cie, BP 3, 78370 Plaisir, France
Stuttgart	Institut für Aerodynamik und Gasdynamik, Universitat Stuttgart, Pfaffenwaldring 21, D7 Stuttgart 80, West Germany
6585 TESTG	6585 Test Group Test Track Division, Holloman Air Force Base, New Mexico 88330, USA
UBCM	Mechanical Engineering Department, The University of British Columbia, Vancouver, Canada
ULC	The City University, Department of Aeronautics, St John's Street, London EC1, England
UKP	Physics Department, University of Kent, Canterbury, Kent CT2 NR, England
ULICA	Imperial College of Science and Technology, Department of Aeronautics, Prince Consort Road, London SW7 2BY, England
ULICME	Imperial College of Science and Technology, Department of Mechanical Engineering, London SW7 2A2, England
ULQMC	Queen Mary College, Department of Aeronautical Engineering, Mile End Road, London E1, England
UOES	Department of Engineering Sciences, University of Oxford, Parks Road, Oxford OX1 3P1, England
USAA	University of Southampton, Department of Aeronautics, Southampton SO9 5NH, England
USME	Mechanical Engineering Department, University of Surrey, Guildford, Surrey, England
UTSI	University of Tennessee Space Institute, Tullahoma, Tennessee 37388, USA
VKI	Von Kármán Institute for Fluid Dynamics, 72 Chaussée de Waterloo, 1640 Rhode-St-Genèsé, Belgium
Volvo	Volvo Flygmotor AB, Aerodynamics Department, S-461 01 Trolhattan, Sweden

VPI	Virginia Polytechnical Institute, Department of Aerospace Engineering, Glacksburg, Virginia 24061, USA
Wash U	University of Washington, Seattle, Washington 98105, USA

APPENDIX 1

MEMBERS OF SUBCOMMITTEE ON WINDTUNNEL TESTING TECHNIQUES, TES

Mr Robert O.Dietz (Chairman)
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NOTE: M. Ph.Poisson-Quinton assisted M. l'Ingenieur Général Carriere in work of TES.
Mr L.H.Ohman assisted Dr Orlik-Rückemann in work of TES.
Mr P.G.Pugh assisted Dr G.G.Pope in work of TES.
Dr W.Lorenz-Meyer assisted Prof. Dr Barche in work of TES.

APPENDIX 2

MEETINGS OF SUBCOMMITTEE ON WINDTUNNEL TESTING TECHNIQUES SINCE MARCH 1975

10 October 1975	London, England
7 May 1976	ISL, Saint-Louis, France
2 October 1976	Ames Research Center, USA
4 May 1977	Copenhagen, Denmark

All meetings were held in connection with the Fluid Dynamics Panel Meetings to avoid travel costs.

APPENDIX 3

LIST OF PERSONS WHO ACTIVELY HELPED IN THE WORK OF TES

Amiet, R. (UTC)
Baals, D. (LaRC)
Bazin, M. (ONERA)
Bardin, M. (ONERA)
Bauligman, R. (GD)
Beamish, G. (GD/FW)
Bechert, W. (DFVLR)
Berndt, S. (FFA)
Berogud (Aerospatiale)
Bhat, W. (Boeing)
Binion, T.W. (AEDC)
Bongrand, J. (CEPr)
Bowker, A. (NAE)
Boyden, R.P. (LaRC)
Brocard, M. (AECMA)
Broderson, D. (MCD)
Brooks, J.R. (RR/BED)
Broussaud, P. (ONERA)
Buckley, J. (LaRC)
Carrara (STAe)
Cheeseman, I.C. (SU)
Chevallier, J.P. (ONERA)
Clapper, B. (GE)
Covedor (AMD/BA)
Covert, G. (MIT)
Cox, M. (NGTE)
DeMetz, F. (NSRDC)
Destuynder, R. (ONERA)
de Wolf, W.G. (NLR)
Dougherty, S. (AEDC)
East, L.F. (RAE)
Endier (ONERA)
Erickson, J.C., Jr (CAL)
Fiore, R.W. (AFFDL)
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Mole, P. (GD)
Morceau (AMD/BA)
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Ritter, A. (CAL)
Robinson, B. (Lockheed)
Ross, R. (NLR)
Sandford, J. (VKI)
Schlinker, B. (UTC)
Schofield, B. (Hawker-Siddeley)
Scott, C. (LaRC)
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Stainback, P.C. (LaRC)
Steinle, F.W. (ARC)
Strout, F. (Boeing)
Summers, W.R. (AEDC)
Tanna, B. (Lockheed)
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APPENDIX 4

AEROACOUSTIC REQUIREMENTS FOR MODEL NOISE EXPERIMENTS IN SUBSONIC WINDTUNNELS

by

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SUMMARY

This appraisal of subsonic windtunnel testing techniques required for aircraft noise-model research supplements the corresponding part of a brief wider review of noise measurement problems in ground-based facilities with forward-speed simulation, issued as Appendix 4 of AGARD-AR-83 about two years ago. In particular, the present discussion covers test-section requirements and circuit design for acoustic tunnels providing quiet anechoic testing environments, special measurement and analysis techniques for noise-model research, and simulation of propulsion noise sources at model-scale. Progress towards the clarification and treatment of some major problem areas is summarised, the principal features and capabilities of available tunnels are listed, and a supplementary bibliography of about 80 directly relevant papers (issued 1975/76) is included. While naturally representing the author's views, this assessment does attempt to reflect the debated experience and expressed opinions of some aeroacoustic specialists from both North America and Europe, who participated in informal two-day 'Workshops' sponsored by the AGARD FDP, at UTRC Hartford (USA) and RAE Farnborough (UK) during April and May 1976.

1. INTRODUCTION

To ensure meaningful evaluation and prediction of flight effects on aircraft noise generation and propagation, reliable representation and measurement of relevant aerodynamic flow conditions as well as of acoustic characteristics must be possible. Acoustic windtunnels, with models mounted in a quiet test-section airstream surrounded by an anechoic working-chamber, have now been established as primary tools for noise-model research work and should next be exploited also for the direct support of specific quiet aircraft projects. The special advantages of such tunnels in ensuring a more sheltered and controlled environment than outdoor mobile-model facilities and flight testing include capability and continuous operation, repeatable test conditions, high productivity, good measurement accuracy, testing flexibility, and the precise alleviation of reflections from neighbouring surfaces. Of course the recent experience on noise testing under forward-speed conditions and on associated techniques is still very limited at both model-scale and full-scale, as compared with extensive and continuous aerodynamic testing over half-a-century. However, most of the problem areas associated with subsonic tunnel design and application for noise-model testing, as identified earlier in Reference 1 (Tables 1 and 2), have now been clarified. Also, the special treatments or limitations involved have become quantifiable in many respects, as discussed later under the convenient main headings of Tunnel test-section requirements (Section 2), Tunnel circuit design (Section 3), Special measurement and analysis techniques (Section 4) and Model-scale simulation (Section 6).

In particular various parasitic noise fields, which can be produced naturally by the testing environment and which could mask the true measurements of model noise (Fig.1) may be precluded or alleviated by applying and extending existing design experience in noise reduction and tunnel airflow control.

(1) Acoustic lining of the working-chamber can minimise reverberation effects down to acceptably low frequencies (Section 2.1); here an open-jet test-section with the working-chamber wall treatment well clear of the airstream offers distinct advantages over a closed test-section.

(2) Acoustic treatment of the tunnel circuit can reduce substantially the intrinsic background noise which could reach the tunnel test-section from the tunnel drive-fan and circuit (Sections 3.1 and 3.2); here a fan of low tip-speed located well remote from the test-section, in a low speed duct region providing a high tunnel contraction-ratio, alleviates the penalty for adequate silencing (Section 3.4).

(3) Good quality mainstream flow into the test-section (Section 2.3) and prevention of significant flow changes with powered-model conditions helps to avoid spurious noise generation; the former requirement tends to favour a closed-circuit tunnel and the latter an open-return.

(4) Model rig and microphone arrangements should be carefully chosen and tailored to reduce their self-noise and local aero-acoustic interference in the airstream (Section 4.2); with an open test-section the far-field measurement microphones and supports can be located outside the test-section airstream, in nominally still air, but the effects of noise propagation across the airstream mixing boundary then have to be allowed for (Section 4.3).

For practical application to full-scale far-field conditions, reliable noise measurements should be achievable within the 'free-field' portion of the model-source far-field, where the sound-pressure-level varies inversely as the square of the distance (spherical radiation) apart from atmospheric attenuation. Since this 'free-field region' is bounded internally by the 'near-field' region of the noise source and externally by the 'reverberation-field' of the working-chamber, the maximum permissible size of model and the minimum permissible size of test-section are restricted from acoustic as well as aerodynamic considerations (Sections 2.1 and 2.2). Also a minimum acceptable size of model can be determined by practical difficulties in achieving adequate microphone response and resolution simultaneously with high frequency measurements, as well as from representative model construction problems at small-scale.

To relate the tunnel model experiments directly to conventional flight conditions, an analytic framework has to be specified for the appropriate frame-of-reference transformations. This must convert from the relative motions for the fixed model in the tunnel airstream with the microphones also fixed inside or outside the airstream, across to the moving aircraft in ambient still air with the microphone fixed on the ground (Figure 2, Seftion 4.1). Current practice is to correct tunnel measurements for the absence of elementary Doppler shift effects on sound frequency, for the presence of elementary airstream convection effects on sound directivity angle, and for simple refraction effects through the airstream mixing boundary if external microphone locations are employed (Figure 3, Section 4.3).

Several acoustic tunnels with equivalent airstream diameters less than 3 m are already in regular operation and others are nearing completion (Section 5.1). However, the need for much greater tunnel size from acoustic as well as aerodynamic constraint considerations has required developments towards the application of some existing large tunnels, such as the RAE 24 ft open-jet and the NASA 40 ft x 80 ft closed-jet (Section 5.2). The experience obtained has naturally stimulated proposals for economic modernisation of these large tunnels, and has influenced recent design specifications for new large tunnels such as the German-Dutch DNW 8 m x 6 m. Furthermore, recent promising developments in directional acoustic receivers together with other noise-discrimination techniques mentioned in References 5 and 7 could ultimately prove sufficiently practical and flexible to allow substantial relaxation of some tunnel acoustic treatment requirements, though the noise field diagnostic capabilities for the model under test could then be correspondingly poorer or more complex. Thus, some current restrictions on noise-model testing in conventional tunnels could thereby be alleviated, such as allowing otherwise objectionable levels of tunnel background noise or semi-reverberant closed test-sections.

For adequate simulation at model-scale in tunnels or other facilities, the relevant geometrical and constructional features have to be selected for representation in relation to any aerodynamic, elastic and dynamic aspects particularly affecting noise generation, with overall consideration of scaling factor implications. Some non-dimensional similarity parameters then have to attain values at model scale reasonably close to those of interest full-scale; e.g. Mach number, Reynolds number, and effective-speed ratios (blade-tip/airstream, or engine-flow/airstream). Essentially, shortfall in some of the parameter values may have to be accepted in practice as of secondary importance, and interpreted in the light of experimental variations of the parameter values and other experience. At the same time, other scaling factors such as selected Strouhal number (frequency parameter) and sound-level coefficients should be validated experimentally as applicable to full-scale practical prediction for the particular aircraft type of interest, for example by comparisons at different model scales. Of major significance is the meaningful representation (qualitative and quantitative) of the full-scale noise sources and radiation characteristics from propulsive systems and adjacent surfaces, or from any powered-lift schemes. Fortunately, with careful appreciation of the specific research task, only partial simulation of the engine noise sources and engine airflow is needed for studies of the particular noise changes due to forward speed. Even so there are significant problems including model-drive and model-support implications (Section 6).

2. TUNNEL TEST-SECTION REQUIREMENTS

2.1 Measurement Frequency Range and Model-Scale

Conventional absorber techniques can be employed in the anechoic design and application of windtunnel test-sections for aircraft noise research, though a variety of special testing requirements can then arise as discussed throughout this paper. As a rough working rule for the acoustic treatment of tunnel test-section boundaries, adequate absorption of incident sound energy can be achieved by foam sheet covering (thickness t) for wavelengths up to $\lambda_{max} \approx 2t$, or by foam wedges (height h) up to $\lambda_{max} \approx 5h$. Nevertheless in practice there can be significant regions which are not amenable to appropriate acoustic treatment for aerodynamic or structural reasons, including downstream or upstream facing areas in or at the ends of the test-section leg. For open-jet tunnels, an adequately anechoic test-section can be provided in principle without appreciable aerodynamic interference on the test-section airflow, simply by appropriate acoustic lining of the large working-chamber well clear of the open-jet boundaries. But acceptable treatment of the collector entry and of any supporting structure for the jet nozzle can be difficult. For closed-jet tunnels, the acoustic lining of the test-section tends to be more limited because the surface presented to the airflow has to be relatively smooth, streamlined and hard-wearing. The outer covering should preclude objectionable aerodynamic interference with the test-section airflow, or the generation of additional airflow noise, while not impairing the sound absorption efficiency of the particular scheme nor allowing deterioration due to long-period scrubbing effects.

The full-scale frequency range (F_{min} to F_{max}) of subjective interest for the prediction of perceived noise levels is typically from 50 Hz to 10 kHz. The lower limit on measurement frequency in model tests (f_{min}) is usually prescribed

by the increasing difficulty of providing an adequately anechoic test-chamber at lower frequencies, though the problems are more tractable with an open test-section (cf closed) since the acoustic treatment of the solid boundaries (walls) is then far-removed from the test-section airflow. The upper frequency limit in model tests (f_{max}) is usually determined by the reductions in microphone size necessary to ensure adequate frequency response and spatial resolution at the higher frequencies, though with reduced signal strength, and by the problems of adequate allowance for atmospheric attenuation and directivity corrections with the higher frequency. Hence, to ensure an adequate frequency range at model scale (length l) of subjective interest at full-scale (length L), for appropriate correspondence at the same Mach number and same Strouhal number, the permissible model size is correspondingly limited in that

$$F_{min}/f_{min} > l/L > F_{max}/f_{max} .$$

Thus typically, with at best $f_{min} = 200$ hz and $f_{max} = 80$ kHz ,

$$1/4 > l/L > 1/8 .$$

Such arguments are not intended to decry the usefulness of smaller-scale or larger-scale models, particularly over more limited ranges of frequency. But they serve to stress the importance of matching the model size and measurement frequency range to the capabilities of the particular anechoic chamber and of the available instrumentation, and the necessity for improvements in relevant instrumentation capabilities. More generally, the model size has to be made compatible also on a variety of other aero-acoustic counts, as discussed in the following sections.

2.2 Acoustic Wavelength and Geometrical Constraints

The extent of the near-field region from the noise model depends in general on the noise source type (monopole, dipole, quadrupole) and the intensity. But, for a compact source, it is roughly of th order of one or two wavelengths. Thus, to ensure that the acoustic far-field noise conditions (spherical radiation) are attained within the test-section air-stream (radius R_{air}), the latter must extend to say at least 1.5 times the maximum wavelength λ_{max} ($\equiv a/f_{min}$) of interest from the model noise source. Moreover, to provide measurement conditions free of the boundary near-field interference effects, the measurement points should be located at least a distance (B_{mic}) of say $0.3\,\lambda_{max}$ from any acoustically-treated wall or airstream 'free-jet' mixing boundary. Hence, as illustrated diagrammatically by Figure 5, for a centrally located compact noise source such accoustic wavelength constraints imply

$$R_{air} > 1.5 \times (a/f_{min}) \quad \text{and} \quad B_{mic} > 0.3\,(a/f_{min}) .$$

The advantages of employing a large test-section are clearly evident from this aspect of permitting adequately long wavelengths (low frequencies), appropriate to large model size.

With practical models, as distinct from single compact noise sources, the finite geometry and character of the spatially large distribution of noise source elements across and along the tunnel airstream need to be allowed for to ensure attainment of far-field measurements. For then, the extent of the near-field of the distributed noise source depends strongly not only on the wavelength (or frequency) of interest, but also on the relevant characteristic dimension of the noise source and the possible variation of predominant frequency and sound power along its extent. Indeed, the choice of characteristic dimension itself could vary with the frequency band and noise measurement direction of primary concern. Such model 'geometric' size constraints can be more significant than the 'wavelength' constraints of the preceding section. The formal specification of general working rules for predicting the near field limits for practical noise models covering our interests is thus still difficult.

For an open-jet tunnel, with the test-section airstream surrounded by a much larger anechoic chamber, it could be argued that full development to such geometric far-field conditions need not be achieved until well outside the airstream jet-mixing boundary. Then, because of the alleviation on the foregoing constraint arguments, relatively larger models might be permitted; assuming of course that other acoustic wavelength constraints, aerodynamic constraints, and avoidance of distortion of the airstream boundary are not already limiting factors. Nevertheless, the use of microphone locations outside the airstream mixing boundary can introduce doubts concerning noise propagation characteristics across the varied and complex flow field between the model source and the microphone (Section 4.3), particularly if the source-noise characteristics are unknown or varied. Any increase in source signal strength from a larger model tends to be counterbalanced by the attenuation effects from the more distant measurement points required.

An important example arises in the application of a small 'free-jet' tunnel for testing of a model-jet coaxially-centered on the mainstream jet, with the need to achieve a model-scale as large as possible without excessive testing constraints. Here, from turbulent jet-flow development concepts and practical experience, it can be argued that the aero-acoustic interference of the mainstream-jet development on the model-jet source-noise generation is only negligible if the ratio of mainstream-jet diameter to model-jet diameter is at least 10. However, unless this diameter ratio is even much greater (> 50 say), far-field measurement will necessitate microphone locations well outside the mainstream rather than within. Then the noise-propagation corrections associated with refraction effects through the mainstream external mixing boundary can be substantial for realistic Mach numbers; see Section 3.4 and Figure 4. Additionally the problem becomes much more complex and the available corrections more questionable if the model-jet is inclined to the mainstream or off-centre, or if airframe installation/interaction effects are to be explored.

Thus, in any practical noise experiments, it is advisable to explore the sound field at different distances as well as different directions from the model, in order to establish that adequate far-field conditions have been reached at the measurement points to the standard of accuracy required. More specifically, further quantification of the type of constraints raised in this and the preceding section could now profitably follow from a declaration and critical analysis of relevant experimental and theoretical results, supplemented by some specially directed and carefully controlled explorations of noise fields during future model testing programmes in acoustically-treated tunnels. There is an urgent need for such reliable guide-lines to expedite more profitably designs of models, facilities and experiments for investigating forward-speed effects on noise. Even static test results for elementary models (if precise) could help the formulation of useful working limits for far-field measurement locations under forward-speed conditions in the light also of reasonable theoretical concepts; typically from diagnostic field status on small-scale models in large anechoic chambers.

2.3 Flow Quality Requirements

The desirability of good uniformity, steadiness, and low turbulence of the flow in the test-section airstream is already well established for aerodynamic-model testing. The possible significance of such flow quality considerations on noise-model testing, either directly or indirectly through the influence of resulting aerodynamic changes on model noise generation and propagation characteristics, still needs to be clarified and quantified. In particular, there appears to be little quantitative appreciation as yet of the influence of intensity and scale of the turbulence in the oncoming main-stream as regards noise generation at the model, except that the influence could be relatively small perhaps for a jet efflux but significant for a fan intake. Nevertheless, these aspects certainly cannot be ignored, not merely for fan-model noise, but also for airframe-model noise and engine installation effects, at least for small-scale models; i.e. when the aerodynamic flow field under the low Reynolds number conditions can vary appreciably with stream turbulence. The declaration and analysis of any existing relevant results is now badly needed, complemented by some exploratory noise measurements and related aerodynamic studies in existing tunnels with known variation of turbulence, particularly on fan-powered models.

Noise measurements employing a microphone inside the tunnel airstream (rather than outside) are frequently required for far-field studies as well as near-field, unavoidably so with closed-jet tunnels and usually with large open-jet tunnels (Section 4.2). However, with a very quiet tunnel the background noise, as measured inside the airstream by a microphone even when fitted with a nose-cone and pointed directly upstream, can still be largely due to the interaction of the airflow with the microphone rather than the true quiet-tunnel noise levels. More specifically theoretical arguments suggest that, if u' is the rms longitudinal velocity fluctuation and U the airflow mean velocity, then the rms momentum-pressure fluctuation p_H' associated with the turbulence should be $\rho U u'$, while the static-pressure fluctuation should be about $\frac{1}{2}\rho u'^2$. Preliminary RAE experimental results, for $20 \log (p_H'/\rho U u')$ as a function of Strouhal number fD/U where here D is the microphone nominal diameter, confirm that the microphone alone (without nose-cone) measured $\rho U u'$ as expected over a major portion of the Strouhal number range; the fall-off in microphone response at high frequency is also in line with the reduction in scale of turbulence $(L \approx U/2\pi f)$, as expected. However, when a nose-cone is fitted, the microphone does not measure $\frac{1}{2}\rho u'^2$ but a fraction of $\rho U u'$ depending on the Strouhal number, this fraction being dependent on nose-cone shape. Admittedly this apparent primary dependence on $\rho U u'$ does not in itself provide a clear physical explanation of the microphone response to turbulence, particularly since the spectra of lateral component v' are similar to those of u' in the experiments presently completed. Nevertheless, the practical significance of these results cannot be ignored in that turbulence levels below 0.1% seem essential for acoustic tunnels, if the turbulence induced signal at the microphone (with nose-cone) is to lie below the intrinsic low background noise of the tunnel.

Now for the practical achievement of high quality airflow in tunnel test-sections, closed-return circuit designs are usually preferred to open-return (straight-through) designs. Additionally, the closed circuit helps to isolate the tunnel testing from the nearby outside environment, thus precluding spurious changes in model test conditions due to ambient winds and external noise, while partially shielding the surrounding community from objectionable testing noise. However, for the avoidance of spurious noise generation by models (again particularly fans), it is equally important to ensure that no significant distortion of the test-section airstream can arise from possible persistence of the model wake/efflux or from circuit-flow intereference round the closed-return. Fortunately modern circuit designs provide appreciable distance for the jet efflux to disperse before reaching the tunnel fan remote from the test-section (not immediately downstream), which should reduce recirculation effects; also slotting of the collector of open-jet tunnels should help reduce flow unsteadiness. Nevertheless, further tests are desirable to quantify the effects of high energy efflux inserted in the test-section, particularly if directed at a large angle to the airstream direction or well off-centre, when more severe distortion may make it necessary to devise a scheme to more rapidly diffuse or even remove the jet efflux.

3. TUNNEL CIRCUIT DESIGN

3.1 Background Noise Generation

Typically, the background noise level in the tunnel test-section or working-chamber must be 10 dB or more below the model source noise, over the frequency range of interest at the measurement points, to ensure adequate resolution (within $\frac{1}{2}$ dB) of broadband spectra without reliance on special discriminatory techniques. To ensure that noise spectra and directivity patterns at model-scale can be reliably extrapolated to full-scale flight, the tunnel usable speeds should

approach closely those for take-off and landing, e.g. at least 50 m/s (100 kn) and preferably up to 100 m/s (200 kn). Large-scale models are of course required for good aero-acoustic similarity and to preclude microphone measurement problems at high frequencies, but tunnel background noise tends to increase at lower frequencies and anechoic chamber demands become more difficult. Furthermore, it is worth recalling that the source noise levels available for measurement at acceptable microphone locations may not increase with greater model-scale, if far-field limitations at the correspondingly lower frequencies for similarity necessitate also correspondingly greater microphone distance from the source. The principal factors contributing to the background noise are included in Figure 1 as part of the interacting acoustic aerodynamic elements associated with model noise measurement in windtunnels.

External ambient noise effects may warrant particular consideration in the design of open-return (straight-through) tunnels and for test-sections not protected by an acoustically-treated working-chamber. Structural transmission of mechanical vibration and motor noise from the tunnel drive system may require special precautions, but problems can be avoided by heavily constructed and damped components with appropriate isolation joints. Minimisation of model rig noise may need special attention when air has to be supplied to model jets and fans, or to resonance-type generators, since internal airflow noise from valves and pipework has to be avoided, along with externally-generated noise from possible vortex shedding and other aerodynamic interference by supporting structure/wires. Spurious noise can likewise be generated by measurement devices and their supports located in the airstream, but the influence of turbulent airstream pressure fluctuations on the noise recorded by the microphone (Section 2.3) or of microphone support vibrations tends to be of more practical concern.

The residual background noise elements, apart also from possible working-chamber boundary constraints on model noise propagation (Section 2.2), may be considered to make up the intrinsic background noise associated with the tunnel-fan and circuit aerodynamics. In general, the design characteristics needed for a good aerodynamic tunnel with uniform low-turbulence flow in the test-section tend also to help towards providing a quiet tunnel by minimising unsteady separated flow conditions around the circuit and by careful aerodynamic design of the fan-in-duct combination. Again, for aerodynamic reasons, anti-turbulence screens and honeycombs are usually located in low-airspeed regions, so they need not create any significant self-noise problems with a reasonable tunnel contraction-ratio (say > 6). However, the possibility of embarrassing self-noise generation by other tunnel flow-control devices must be kept in mind; for example, essential turning vanes and support struts/wires in the circuit flow must be designed or damped to avoid intrusive noise due to 'singing'. Equally well, any inserts for acoustic treatment should neither promote significant self-noise in the flow (Section 3.4), nor introduce troublesome wakes.

In respect of choice of test-section type, with either free or walled boundaries at the edges of the airstream, the open-jet at first sight would appear the more attractive for low background noise levels. The noise emanating from the contraction nozzle and the collector/diffuser can then radiate freely (at least hemispherically out of the test-section) along with that from the model under test, without significant reflection from the acoustically-treated distant boundaries of the surrounding large working-chamber. In principle, the achievable lower limit to background noise may be expected to be set by the broadband quadrupole-type noise produced by the turbulent mixing at the free-jet boundary, for measurement points well within the airstream or several diameters outside. However, a special feature for most open-jet tunnels is the apparent need for 'tabs' protruding from the jet-nozzle periphery into the airstream, and/or for venting of the collector by a cowl or wall slots, in order to preclude possible mainstream jet instability and low-frequency unsteadiness over the operational speed range (Section 3.3). For very quiet tunnels, aeroacoustic problems may then include possible excess noise and jet-boundary thickening from such tabs, collector noise from jet impingement and its possible variation with aerodynamic model testing condition, and adequate sound absorption treatment of the collector-entry/cowl still satisfying aerodynamic and structural needs. Correspondingly for closed test-sections, excess noise can be generated by the high-speed airflow over the test-section walls, especially with acceptable acoustic lining which itself may be of limited effectiveness because of other aerodynamic constraints, while other spurious noise and aero-acoustic constraints can be associated with the essential location of even the far-field microphones in the airstream.

The tunnel airflow-drive represents of course the primary source of background noise in the test-section, unless especially designed to have low noise characteristics (Section 3.2), and located far enough away from the test-section that sufficient circuit length is available in-between to permit adequate in-duct sound-absorption treatment (Section 3.4), with tolerable aerodynamic performance penalties. Assuming dipole-type sources to be predominant, the overall sound pressure level of the fan-generated noise increases roughly as V_T^6. Then, in broad terms, each doubling (or halving) of tunnel speed corresponds to an increase (or decrease) of background noise by about 18 dB.

As a convenient measure of the low levels of tunnel background noise which can be achieved in practice, the sound pressure levels for the successful small acoustic tunnels built at UTRC (effective jet diam 0.7 m) and NSRDC (2.4 m) are as much as 40 dB lower than for the existing untreated tunnels, when compared at the same speed and at the same Strouhal number. Also, as regards acceptable standards of background noise levels, representative powered models suitable for far-field noise experiments in such acoustic tunnels can usefully be tested at airspeeds up to 50 m/s at least, without significant background noise problems and without the need for discriminatory techniques. However, spurious noise generated by the model-rig and in-flow microphone support (unless properly streamlined) can noticeably augment the intrinsic background noise of the quiet tunnel and also exceed the airframe self-noise from a *clean* unpowered model, all usually rising together with increase in airspeed, whereas the model powerplant noise may simultaneously decrease.

3.2 Tunnel Airflow Drive

For subsonic tunnels, the conventional fan system with its well-developed continuously-running capabilities still tends to be preferred for the tunnel airflow drive; usually of an axial-fan type, though not always so for straight-through tunnels where other extractor-type fans can be conveniently exploited. The often contemplated air-injector drive may appear simpler than the fan drive, and probably cheaper if appropriate compressed-air supplies are already available on site. But, even if this pressure/induction system may be made as acceptably quiet as the fan drive, the limitations on running periods then available would often be unacceptable for general low-speed testing.

The tunnel fan itself can contribute a major component of the background noise level in the test-section, especially at low frequencies. Consequently the fan design and the duct length available for acoustic treatment between the fan location and the test-section, both represent critical features as regards background noise limitations and acceptable airspeeds for any noise-model testing. The tunnel fan noise is mainly identifiable as of broadband dipole-type, usually attributed to lift fluctuations on the blades and associated with vortex shedding at the trailing-edges. Typical experimental results are consistent with this, in that the fan sound pressure level tends to increase as the fifth to sixth power of the fan rotational speed. A useful crude working rule when comparing fans under similar operating conditions is:–

$$\text{(Fan overall sound power)} \propto (\text{Tip speed})^3 \times (\text{Aerodynamic Shaft-Power}) \times (1 - \eta)\,;$$

where, providing the fan aerodynamic efficiency η [$\equiv$(pressure-rise power)/(shaft power)] is known, the inclusion of the aerodynamic power dissipation factor $(1 - \eta)$ offers a reasonable basis for comparing the noise of fans of differing design, since the aerodynamic losses and noise generation are closely related. The spectrum at a given rotational speed has a decreasing sound pressure level with rising frequency, but with discrete tones superimposed at the blade passing frequency and harmonics, whose intensities can be a function of inflow turbulence as well as tip-speed. Increase in inflow turbulence to the fan can also aggravate the broad-band noise.

For low noise, the fan should be designed to operate near the condition of maximum aerodynamic efficiency, avoiding significant flow separation regions on blades, but with a low tip-speed – say about one-third the speed of sound. Nevertheless, it must be appreciated that there can be a significant variation of airstream total head across the fan entry section, remaining sensibly axisymmetric if due to boundary effects in the test-section and duct upstream of the fan, but possibly with deviations due to non-axisymmetric acoustic treatments of the upstream ducts. Fortunately the well-designed single fan has proved a powerful and accommodating tool for providing a uniform distribution downstream of the drive section. With modern aerodynamic design methods, essentially involving the choice of an appropriate blade-twist distribution from fan-hub to tip for the expected velocity distribution into the fan and the required pressure rise through the fan, only small adjustments should have to be made to the predicted fan design after appropriate model fan-in-duct checks. Broadly speaking, a tunnel design which incorporates large contraction-ratio is favourable to low noise, because of resulting reductions in both the aerodynamic power required and fan tip speed, for a prescribed test-section size and speed. Now that several acoustic tunnel fans have been built to various quiet designs (Section 5), experimental results on their aero-acoustic performance must be critically evaluated and compared for future guidance, before the construction of new larger-tunnels or of new quiet fans for existing large tunnels. More generally, although continuous variation of tunnel speed is usually achieved through alteration of fan rpm, the fan aerodynamic efficiency and quietness could usefully be further improved (especially for large blockage changes) by incorporation of adjustable or variable blade angle, possibly even with some facility to adjust blade twist or effective camber at least during the installation proving stage.

3.3 Open-Jet Nozzle and Collector-Flow Interactions

In most subsonic tunnels, either with open or closed test-sections, careful tailoring of the test-section design as well as the tunnel circuit and airflow drive is invariably needed to ensure that the test-section flow is steady throughout the required airspeed range. Often, any shortfall or improvement in this respect is manifest also in the degree to which the allied requirements of minimum pressure gradients and minimum energy losses are met. Here, we are primarily concerned about possible low frequency unsteadiness (pulsing) in open-jet tunnels, largely associated with interaction between the jet nozzle flow and the collector, which could generate excessive background noise and intolerable flow conditions. For preciseness, the primary origins of such flow unsteadiness (and appropriate treatments) can be divided into two different categories, though these can arise simultaneously.

Firstly, mixing at the tunnel airstream boundary, while traversing the space between the nozzle and the collector, results in entrainment of the order of 10% excess volume flow which has to be split at the collector entry. With the large-scale eddy sizes arising in this mixing, the instantaneous quantity to be split will show appreciable amplitude variations in the very low frequency range, so that tunnels with simple bellmouth collectors have experienced low-frequency variations in tunnel airspeed apparently associated with this unsteady entry flow. All modern open-jet tunnels with closed-return circuits incorporate some form of ventilation slots downstream of the collector to accommodate this variable spillage and attenuate pressure waves which might otherwise propagate round the tunnel; in some cases extensive ad-hoc tailoring of the slots has been required to obtain satisfactory tunnel flow, e.g. DFVLR Porz-Wahn and VKI, while in others a single peripheral slot has given satisfactory results. Clearly, the aerodynamic design of the collector cowl has to be carefully tailored to the particular test-section and working-chamber configuration, while the cowl must also have acceptable structural and acoustic characteristics.

Secondly, nozzle/collector-edge-tones and related jet-flow oscillations are usually considered to originate from aerodynamic resonance between a disturbance (e.g. ring vortex) leaving the jet nozzle, impinging on the collector cowl, and then feeding back a new disturbance which arrives back at the nozzle in phase with the creation of another disturbance at the nozzle. The frequency of such edge-tones, (primary and higher-order) tend to increase with greater mean airspeed at the jet nozzle and then decrease with greater separation distance between the jet nozzle and collector. In some closed return-tunnels, severe vibrations of the tunnel structure have arisen when the jet/collector edge-tone frequency coincides with the organ-pipe resonance frequency of the tunnel duct. Two very different designs of windtunnels, the RAE 24 ft tunnel (also a fifth-scale model) with 3.5:1 contraction-ratio and the DFVLR Porz-Wahn tunnel with 10:1 contraction ratio experienced these severe organ-pipe resonances; while other tunnels of intermediate design, e.g. the model of the new DNW tunnel with 9:1 contraction ratio, apparently show no signs of this phenomenon. Fortunately when it does occur, this type of aerodynamic resonance can be readily suppressed or reduced by the introduction of peripheral tabs in the form of spoilers or discrete vortex generators at the nozzle outlet, to preclude regular formation of the jet ring vortices; but there can be some penalty because of possible increases in high frequency noise. It should be noted that in some cases (e.g. DFVLR Porz-Wahn) venting of the collector alone produced no noticeable attenuation of these organ pipe resonances.

Additionally, with the introduction of sealed anechoic chambers surrounding an open test-section, an alternative type of edge-tone resonance appears possible involving low-frequency standing waves in the chamber, rather than a return-circuit organ-pipe resonance. This supplementary type was apparently presented in the UTRC Acoustic Tunnel (open return circuit), there again cured by the use of peripheral tabs, and was probably responsible for initial resonance problems in the NSRDC tunnel but there cured by collector slotting.

In view of the wide variations in severity of the unsteadiness problems reported in different open-jet tunnels, and the large variety of collector cowl shapes and venting configurations employed, more basic research would still seem worthwhile to further clarify the fundamental causes of the various types of unsteady phenomena and to provide detailed guidance for their avoidance in acoustic tunnels with minimum penalties in other respects. More generally, for dynamic as well as noise testing in open-jet tunnels, it seems essential to establish whether the test-section airflow can really be guaranteed to be as steady as that in good closed test-section tunnels, or whether some low-frequency unsteadiness and relatively higher levels of turbulence will remain despite jet-nozzle and collector treatments. The aerodynamic and acoustic significance of deflection of the open-jet boundary due to the presence of lifting models, particularly with powered high-lift systems, needs also to be explored further.

3.4 Noise Absorption Treatment of Tunnel Circuit and Aerodynamic Implications

Significant unknowns and restraints can arise in attempts to apply, efficiently and economically, sound-absorption techniques to substantially reduce tunnel background noise; by internal-circuit treatments between the drive-fan and the test-section. Some compromises in wall-lining and splitter designs are essential because of the following factors. Broadly speaking the absorption of high-frequency noise requires closely-spaced splitters, whereas low-frequency absorption demands greater lengths. Local airspeeds and airstream-pressure losses tend to rise with silencing efficiency over a wide range of frequency, because of the more extensive circuit-flow blockage and larger wetted areas, even with careful streamlining. Low airstream-pressure losses are needed to ensure high airspeeds in the test-section with acceptable power and cooling requirements. Objectionable self-noise and reductions in absorption efficiency can be caused by high speed airflow over the absorber surface areas. Reductions of absorber efficiency may be associated with the needs for surface protective covering and structural integrity in high-speed airflows without costly maintenance over a period of several years. Such considerations suggest that the most favourable tunnel-circuit locations for the application of sound-absorption techniques are where the airspeed is near the minimum, i.e. in the settling chamber upstream of the contraction and after considerable diffusion well downstream of the test-section, commensurate of course with the fan location, and with the circuit type and geometry.

Thus the elaborate NSRDC tunnel (Fig.6) incorporate long 'acoustic mufflers' in the especially large legs of the closed return circuit immediately upstream and downstream of the quiet axial-fan drive, to reduce fan noise reaching the test-section particularly in the low-frequency range but with only minor aerodynamic penalties. Each muffler comprises two sinuous absorptive splitters mounted vertically in the middle of the tunnel and one along each side-wall; the sinuous bends have a large radius to avoid flow separation, while providing additional high-frequency noise reduction by eliminating an unobstructed linear sound path through the muffler, and also increasing the effective length of the passage for a given geometric length of muffler. The aerodynamic total-head losses for each muffler were only about 15% of the overall loss round the tunnel circuit and about the same as the loss through the cooler or through the anti-turbulence screen section. Moreover, the perforated metal coverings of the absorptive material here causes no troublesome self-noise because the duct airspeed at these muffler locations is so low.

The simpler and smaller UTRC tunnel (Fig.7), with its open-return design and extractor-type centrifugal fan at the exit, avoids the need for acoustic treatment of the tunnel inlet upstream of the test-section but assuming of course the absence of any external noise problems. Downstream of the test-section, at the end of the conventional straight diffuser and just ahead of the drive-fan, the tunnel circuit incorporates a Z-shaped muffler (absorptive and reactive) comprising two arrays of parallel baffles/splitters and two lined 90°-bends, which serve to attentuate the fan noise by at least 50 dB for frequencies down to 250 Hz in the test-section. Turning vanes were here not installed in the bends, to preclude the possibility of discrete frequency noise due to 'singing' and of broadband noise generation due to their immersion in turbulent flow from the diffuser.

However, for acoustic treatment of existing aerodynamic tunnels, or for the design of new dual-purpose tunnels where aerodynamic testing still has greater priority than noise testing, other conflicting technical aspects and economic constraints can be limiting factors. The recommended practical modification of the RAE 24 ft tunnel (Fig.8), maintaining the large test-section size (7.3 m diam) with the low contraction ratio (3.5/1), has uniform blocks of low-frequency and high-frequency splitters as part of the multi-passage diffuser downstream of the new fan, with similar blocks incorporated in the first diffuser downstream of the collector primarily because a sufficient straight-length (upstream of the fan) is available only there. Then, since the circuit airspeed is high in the first diffuser, splitter self-noise becomes a serious design factor as well as splitter aerodynamic drag. The possibility of making the splitters sinusoidal along their length to achieve increased sound attentuation by elimination of 'line-of-sight' through the block is precluded here by lack of length when adapting this existing facility. Other designs for the modified RAE 24 ft and for the new DNW 8 m x 6 m tunnel, providing a worthwhile increase in contraction-ratio and thereby improvements in top-speed and flow quality, inherently reduce the mean duct speed and required fan tip-speed for a prescribed test-section speed. So fan-generated noise and splitter self-noise tend to be correspondingly lower, alleviating the acoustic absorption required to achieve the prescribed background noise in the test section. It also becomes more attractive to take full advantage of the changes in direction through the circuit corners, by incorporating absorptive lining in the local wall surfaces and corner vanes.

Estimates of the acoustic properties of feasible splitter arrangements can be attempted using developments of Kremer's theory reported by Beranek and Schultz[85], where the splitters are considered to be of homogeneous porous material, with the acoustical impedance assumed to be a unique function of the through-flow resistance R_1 (Rayls/m) of the material. With the splitter thickness defined as 2d (m) and the gap as 2h (m) between the splitters, the maximum attentuation per unit length is then attained at a particular frequency $f_{opt} = 101.6/\sqrt{hd}$ (Hz), using an absorbent material with the 'optimum' flow resistance $R_{opt} = 667.5\sqrt{h/d^3}$ (Rayls/m). This peak optimum design restricts significant noise attenuation to only a narrow band of frequencies, but attenuation over a much wider band of frequencies can be obtained at the expense of a lower value of peak attenuation, if material with a higher flow resistance is chosen. Of course maximum attenuation for a given splitter length is obtained by making h small, but the airflow blockage can produce excessive drag losses, so that $d \not> h$ proves an essential compromise between the acoustic and aerodynamic requirements, with some lengthening of the splitters to compensate for the reduced attenuation per unit length. Some RAE experimental results showed that peak attenuation occurred near the predicted frequency, but that the peak was substantially less than the theoretical estimates when $R_1 \approx R_{opt}$, while at the lower frequencies the attenuation was greater than estimates. More generally, available theoretical methods for the estimation of noise-absorption and self-noise of splitters with practical surface protection in an airstream still seem inadequate.

4. MEASUREMENT AND ANALYSIS TECHNIQUES

4.1 Relative-Motion Considerations for Tunnel-Flight Correspondence

There exists an elementary direct correspondence between noise measurements for the model fixed in an ideal tunnel airstream (effectively uniform and unbounded) and for the same model in steady level flight at the same relative velocity to the likewise ideal still air, choosing for comparative purposes here a frame of reference fixed to a noise-model (Fig.2). Naturally for such ideal model test conditions in tunnel and flight, with the identical relative airflow velocity, measurements can directly correspond at the same microphone distance from the model, for the same sound emission angle θ (i.e. wavefront-normal inclination) to the direction of relative motion; there is then of course identity also of the 'retarded-time' from pulse emission at the model source to pulse reception at the microphone location. This careful distinction here between correlation of results from different test methods at the same value of the emission angle θ rather than of the 'convected' ray angle ψ, while seemingly trivial at first sight, becomes of vital practical significance for the meaningful comparison and physical interpretation of results at different flight and/or tunnel Mach numbers M and under static rig conditions. *From simple relative airstream convection arguments*

$$\tan \psi = \tan \theta \, (1 + M \sec \theta)^{-1} .$$

In real flight, this emission angle θ is of course the instantaneous line-of-sight angle from the conventional stationary observer to the aircraft (i.e. flight model) at the sound pulse emission time; as distinct from the 'convected' ray angle ψ which represents the instantaneous line-of-sight angle at the corresponding pulse reception time, varying with M even for constant source emission characteristics. In real tunnel tests, if desired, to ensure measurements for unchanged values of emission angle θ as tunnel airspeed is varied (including static conditions), the datum microphone locations (ψ values for assigned θ values) can be displaced geometrically downstream with increasing airstream Mach number, according to the foregoing convection relationship. With an extensive distributed source (e.g. jet efflux) the geometric far-field conditions may not be adequately achieved for the allowable microphone distance from the model, when strictly these angles should be related to several prescribed source elements in turn rather than to the model geometrical location. Even some rough checks, with simple source distributions based on theoretical arguments and other experiments (e.g. static), could usefully indicate the magnitude of possible errors due to the more convenient assumption of a single compact source at or near the model; see also Section 2.2.

To fully complete the practical equivalence (Fig.2), the microphone should not only occupy the same position relative to the model frame-of-reference at pulse reception time (i.e. identical θ), but ideally the velocity of the

microphone relative to the model should also be unchanged. Now in principle, for the same θ, the acoustic pressure amplitude measured by a moving microphone is independent of its velocity relative to the source, though any pulse is then detected over a time period proportional to $(1 + M \cos\theta)$. Hence in practical terms, the stationary microphone with stationary model in the tunnel airstream measures the same proportional-bandwidth mean-square pressures (e.g. $\frac{1}{3}$-octave) as the stationary microphone (conventional observer) with flight model (moving aircraft); strictly provided the tunnel airstream and flight Mach numbers are identical, for the same values of θ, and at the same microphone distance. Even so the tunnel-model frequencies then need to be multiplied by the Doppler factor

$$(1 + M\cos\theta)^{-1}$$

to convert to the flight-model observer conditions with the separation speed Ma. More generally, to allow for essential differences in practice between the tunnel model microphone and the flight model observer distances, it is customary to appeal of course to the far-field inverse-square law, with $\Delta SPL \approx -6$ dB per doubling of distance. Additionally, conventional corrections to allow for atmospheric attenuation may be applied, typically

$$\Delta SPL \approx f/1000 \text{ dB per 150 m distance .}$$

4.2 Microphones Inside Tunnel Airstream

A preference of microphone locations well inside the test-section airstream of a large open-jet tunnel, as well as inevitably for closed-jet tunnels, can follow from the need to make noise measurements as close to the noise source as far-field requirements may permit (Sections 2.1 and 2.2), or as near-field studies will require. Simultaneously, the effects of parasitic flow fields at or outside the mainstream jet boundary, i.e. other than those flows properly associated with the model condition, are then avoided on the model noise propagation characteristics. To reduce the wind-generated noise at the microphone located within the tunnel airstream, for much higher airspeeds than conventional ambient-wind conditions, a streamlined nose-cone with a circumferential axi-symmetric strip of fine wire mesh is usually substituted for the standard flat grid protecting the microphone diaphragm, replacing also the conventional spherical porous windscreen. The nose-cone and hence the axis of the microphone diaphragm are pointed directly upstream at any microphone location to minimise airflow disturbances. The microphone response corrections needed to give true free-field conditions are a function of both the sound frequency and sound incidence at the microphone. Additional free-field corrections for the presence of the nose-cone are often determined by datum microphone measurements made with and without nose-cone, for noise generated by the model at *zero* tunnel airspeed. Fortunately the omni-directional characteristics of the microphone tend to be improved by the addition of the nose-cone, though sound incidence effects are still large at high frequencies and calibration checks are still essential. There is also some justification for the expedient practical assumption that the local airflow over the microphone in the tunnel airstream does not significantly alter the microphone response to the sound received, with of course the nose-cone and streamlined support kept aligned along the local airstream direction. Nevertheless, such microphone response and incidence corrections for in-flow measurements warrant further investigation, particularly with the need for more reliable measurements at higher frequencies.

The future significance of any residual wind-generated noise at microphones located within the airstream of open-jet tunnels or closed-jet tunnels also needs continually to be re-assessed, taking into account possible reductions in (or discrimination from) other parasitic noises, and the signal strengths of future interest from quieter or more complex model-sources. Recent RAE experiments have confirmed that, unless very low turbulence levels are achieved, the pressure fluctuations associated with the turbulence in the airstream can determine the 'apparent' background noise levels of quiet tunnels, as indicated by a microphone (even with nose-cone) in the airstream. Indeed datum microphone measurements inside a very quiet small tunnel, but for very high turbulence levels (airstream u'/U of order 1% rather than conventional 0.1%), recorded SPL values as much as 20 dB higher than measurements well outside the airstream. Some further comments on this aspect can be found in the discussion of 'Tunnel flow quality requirements' (Section 2.3). Some recent advances made in both Europe and North America on the design of microphone probes for industrial sound measurements in turbulent duct flows are of interest; but may not be generally applicable for our purposes where large variations arise in the angular difference between the local-airflow incidence and predominant sound-ray incidence at the probe.

4.3 Microphones Outside Tunnel Airstream

With an open-jet test-section inside a much larger acoustically-treated working-chamber, the microphones for far-field measurements can alternatively be located *external* to the airstream and well clear of the mixing boundary (Fig.3) so as to be in nominally still air; though at a greater distance from the noise source giving a correspondingly weaker signal strength relative to the background noise level. Moreover, possible falsification of the noise measurements needs to be assessed and allowed for, because of the intervention of the mixing boundary between the noise source (within the test-section potential core) and the external microphones.

As regards spurious refraction of sound propagated through the mixing boundary, and even possibly through weaker secondary flow regimes outside the mixing boundary, early studies indicated that such effects were tolerably small and adequately assessable by qualitative ray theory arguments, for the low airstream speeds ($M < 0.15$) then feasible with acceptable background noise levels. More precise theoretical treatments have now been formulated by Amiet[72], Jacques[48] and others, in which the tunnel mixing boundary is modelled simply in terms of a thin vortex sheet of small thickness (compared with the incident-sound wavelength) between the uniform stream and still air without change in density across

the shear layer (Fig.3). Essentially Amiet considers a plane interface and uses ray theory to derive equations which, conveniently for our purpose, permit sound pressure measurements p_m made at an apparent directivity angle θ_m and radius r to be corrected in both intensity and angle for 'ideal tunnel' conditions with the microphone immersed in an infinitely large airstream. The effect of the refraction is not merely to change the ray direction from θ_c' inside to θ_c outside the airstream (Fig.4), but also the intensity through effective changes in ray spreading angle as well as distance. The predicted changes to corrected angle θ_c' from visual measurement angle θ_m , and the corresponding SPL changes $20 \log (p_c/p_m)$ for an equal radial distance r from the source, are certainly no longer small when the airstream Mach number is increased from 0.1 to 0.3 (and to 0.5) for $h/r = 0.15$, where h is the separation distance between the source and shear layer. With increasing M , θ_c' essentially reduces over the whole measurement angle range, while the SPL at equal radius increases over the whole of the forward arc (upstream of source) but decreases over most of the rearward arc. Also, it should be noted that small angles to the airstream direction are not allowable in practice for measurements outside the airflow, because of the rapid variations in the corrections there, both in the forward and rear arcs; in the forward arc even at angles somewhat exceeding those for total internal reflection from the mixing boundary (i.e. even outside the 'zone of silence').

Additionally, the model noise propagation may be subject to frequency and spatial scattering at the test-section mixing boundary, or can augment noise generation from the turbulent mixing itself. Early experiments again implied that the practical effects were small, at least up to the maximum frequency of 10 kHz and for low airstream Mach numbers ($M < 0.15$) then tested. But such scattering effects are envisaged primarily as high frequency phenomena affecting sound propagation at wavelengths less than the turbulence length scales within the mixing region. Indeed, for $M \approx 0.2$, noticeable broadening of a pure tone at about 24 kHz has been displayed by measurements made well outside the mixing boundary of the UTRC tunnel[73], while as much as half the transmitted intensity across a shear layer at very small wavelengths has been attributed to scattered waves from experiments by ONERA[14].

Overall, the complex nature and larger thickness of the mixing boundary needs to be properly appreciated in practical terms, including the influence of any tabs incorporated round the tunnel nozzle periphery to ensure airflow stability, so that representation by simple thin shear layers seems only an expedient gross approximation to the true airflow conditions. At this stage, the available theories and limited measurements should be used for qualitative guidance rather than precise corrections, preferably towards defining the test conditions for acceptably small corrections on the noise changes with forward speed under investigation.

5. SPECIFIC TUNNELS FOR NOISE-MODEL TESTING

5.1 Available Small Tunnels

Within the past five years several acoustic windtunnels have been specially built or existing aerodynamic tunnels correspondingly modified, to provide simultaneously an anechoic working-chamber and low background noise at the test-section with the tunnel in operation. But these existing acoustic tunnels are small with equivalent test-section airstream diameters less than 3 m. The test-section and working-chamber measurements, quoted below in metres, refer either to the diameter x length, or to width x height x length; supplementary dimensions quoted in feet signify only the original designation of the tunnel. Sometimes alternative test-sections may be available, either smaller and faster, both closed and open. The specific references quoted here give more detailed specifications of the particular tunnels. Overall analysis of considerations arising in the design and application of acoustic tunnels is contained in Sections 2 and 3. The provision of very small 'free-jet' tunnels by simple adaptation of anechoic chambers with existing capability of static noise-testing on jets should also be noted, the major jet efflux of largest available diameter then being employed as a mainstream flow, for example about a model jet co-axially centred but of much smaller diameter (see Section 3.2).

American acoustic tunnels already include:

(1) Naval Ship Research and Development Center (NSRDC, 1971)[67]
Open-section 2.4 m diam x 3.5 m; max airspeed 60 m/s, closed-return.
Working-chamber 7.2 m x 7.2 m x 6.3 m; cut-off 150 Hz.

(2) United Technologies Research Center (UTRC, 1971)[73]
Open-section 0.8 m x 0.5 m x 4.8 m; max airspeed 200 m/s; atmos-return.
Working-chamber 6.7 m x 4.9 m x 5.5 m; cut-off 200 Hz.

(3) Massachusetts Institute of Technology (MIT 5 ft x 7 ft modified, 1971)[78]
Open-section 2.3 m x 1.5 m x 2.4 m; max airspeed 35 m/s;
Working-chamber 2.3 m x 1.5 m x 2.4 m; cut-off 600 Hz.

(4) Bolt, Beranek and Newman (BBN, 1975)[84]
Open-section 1.2 m x 1.2 m x 10 m; max airspeed 45 m/s; atmos-return.
Working-chamber 7.0 m x 6.1 m x 13.2 m; cut off 160 Hz.

(5) Lockheed-Georgia (LG, 1975)[62]
Open-section 0.76 m x 1.1 m x 2.9 m; max airspeed 75 m/s; atmos-return.
Working chamber 3.4 m x 5.2 m x 3.4 m; cut-off 200 Hz.

European acoustic tunnels include:

(6) Southampton University (SU 7 ft x 5 ft modified, 1975)[51]
Closed-section 2.1 m x 1.5 m x 4.4 m; max airspeed 30 m/s; closed-return.
Working-chamber as test-section; cut-off 500 Hz.

(7) RAE Farnborough (RAE 5 ft modified, late 1976)[33]
Open-section 1.5 m diam x 2.8 m; max airspeed 65 m/s; closed-return.
Working-chamber 3 m x 3 m x 3 m; cut-off 500 Hz.

(8) CEPr Saclay (CEPr/ONERA, early 1977)[15]
Open-section 2.0 m diam x 9.0 m; max airspeed 100 m/s; atmos-return.
Working-chamber quarter-sphere 9.6 m radius; cut-off 200 Hz.

Other existing aerodynamic tunnels of small-to-medium size have also been given partial acoustic treatment, either around the test-section boundaries or inside the tunnel circuit. These include;

(1) NASA Ames 3.0 m x 2.1 m (7 ft x 10 ft) closed-section.
Test-section acoustic lining.

(2) NASA Lewis 4.6 m x 2.7 m (9 ft x 15 ft) closed-section.
Tunnel circuit acoustic inserts.

(3) Boeing-Seatle 2.7 m x 2.7 m (9 ft x 9 ft) closed-section.
Test-section acoustic lining.

(4) VKI Brussels 3.0 m diam open-section.
Working-chamber acoustic lining.

(5) DFVLR Porz-Wahn 3.3 m x 2.2 m open-section.[27]
Tunnel circuit acoustic inserts.

5.2 Development of Large Tunnels

The only large European tunnel currently incorporating acoustic treatment is the RAE 24 ft tunnel with its open test-section 7.3 m diam x 13 m length, max airspeed 50 m/s, and closed return-circuit. The working-chamber boundaries (13 m x 10 m x 13 m) are now lined with sound-absorbing foam sheet and wedges to provide a cut-off frequency as low as 200 Hz. This 40-year old tunnel has been employed successfully since 1971 for a variety of basic noise-model investigations and for the improvement of associated testing techniques[37,30]. Nevertheless, it is important to stress that the available maximum airspeed of 50 m/s is not used currently for noise-model testing in general, because the tunnel background noise becomes excessive at airspeeds much above 30 m/s. Furthermore, at all airspeeds, the aerodynamic flow quality is considered to be relatively poor by modern tunnel standards, so is unlikely to satisfy future noise research demands. Some acoustic and aerodynamic studies have been made to assess possible practical modifications to the 24 ft tunnel circuit and the 5 ft scale-model[33], in order to double their usable noise-model testing speeds.

In the USA, at least two large aerodynamic tunnels have already been used for noise-model testing. The NASA Langley 30 ft x 60 ft tunnel, with its open elliptic test-section 18 m x 9 m x 17 m length, max airspeed 45 m/s, and closed return-circuit, now has its working-chamber boundaries (34 m x 23 m x 21 m) lined with foam sheet providing a cut-off frequency about 500 Hz (Ref.65). Again, the usable airspeeds for noise-model testing in general reasonably cannot much exceed 30 m/s, while the airflow quality must leave much to be desired in this very old tunnel. The NASA Ames 40 ft x 80 ft tunnel, with its closed test-section 24 m x 12 m x 24 m length, max airspeed 95 m/s, and closed return circuit has rather limited acoustic lining of the test-section boundaries and sometimes none[66]; but special microphone arrays and other schemes are used for some discrimination against reverberant field and background noise levels[59]. Planned improvements to this tunnel (Fig.9) include the installation by 1980 of low-noise fans with much greater drive power to provide the existing test-section with higher max speed (24 m x 12 m, 150 m/s), and the incorporation of an additional circuit leg with much larger test-section (36 m x 24 m, 55 m/s).

More generally, research is now being carried out on the possibility of limited noise-model testing in modern aerodynamic-tunnels, with minimum or no acoustic treatment of their mainly closed test-sections and closed return circuits, but taking advantage of their outstanding airflow qualities (turbulence $u'/U \approx 0.05\%$) and higher maximum airspeeds (> 100 m/s). Experimental investigations, for expediency often in small tunnels at this stage, naturally include the exploitation of directional acoustic receivers and other discrimination/correlation techniques to extract the true source signal from test-section reverberation effects, tunnel/rig background noise, and instrumentation parasitic noise (in the airstream). Ultimately, existing aerodynamic-tunnels with test-section airstream equivalent-diameters of at least 5 m may be profitably employed for some aero-acoustic studies on noise-models using such techniques, appreciating that the latter techniques tend to introduce much greater complexity of measurement, and that other noise-field diagnostic capabilities may be impaired.

Such tunnels could for example include:

(1) ONERA Modane S1 Ma 8 m diam close-section, max airspeed 350 m/s.
(2) RAE Farnborough 5 m x 4.2 m closed-section, max airspeed 110 m/s.
(3) NASA Langley 6.6 m x 4.4 m closed or open-section, max airspeed 100 m/s.
(4) Boeing-Vertol 6.1 m x 6.1 m closed or open-section, max airspeed 130 m/s.
(5) Lockheed-Georgia 7.1 m x 4.9 m closed-section max airspeed 110 m/s.
(6) NAE Ottawa 9.1 m x 9.1 m closed-section, max airspeed 60 m/s.

New large subsonic-tunnels, though intended primarily for improved aerodynamic testing, are clearly also of importance for noise testing. In particular, the DNW German-Dutch windtunnel[28] is now to be built by 1980 at NLR (North Polder), with interchangeable closed and open test-sections 8 m x 6 m x 18 m length, max airspeed 100 m/s, and closed return circuit of contraction-ratio 9/1. For aerodynamic testing, alternative closed test-sections may also be provided, probably 9½ m x 9½ m with max airspeed 55 m/s, and 6 m x 6 m with max airspeed 130 m/s. A 'ventilated' working-chamber acoustically lined will be provided for noise testing with the 8 m x 6 m open test-section configuration. The tunnel is expected to provide a reasonably quiet test-section primarily because the drive-fan has been designed with a lower tip-speed and more moderate aerodynamic loading than previously, taking advantage of the large contraction ratio. But acoustic inserts within the tunnel circuit to further reduce background noise at the test-section will be limited to absorber treatment of the corner vanes, to preclude large power-factor penalties and other constructional difficulties.

The possible 'European low-speed tunnel' studied by the AGARD LaWs Group[1,6], was recommended to have a closed test-section of up to 25 m x 19 m and a maximum airspeed of 130 m/s. From a noise-model testing viewpoint the largest possible atmospheric design, with facility for providing an open test-section surrounded by an anechoic working-chamber, would be preferred rather than a smaller pressurised version and restrictive closed test-section. The tunnel circuit should incorporate a high contraction ratio ($\approx$ 10/1) and a quiet drive, with some internal acoustic treatment both upstream and downstream of the test-section, assuming that the aerodynamic or cost penalties were tolerable. Admittedly, since such a tunnel now seems unlikely to be completed for at least a decade, directional acoustic receivers together with other new discriminatory techniques may then prove sufficiently practical and flexible to allow substantial relaxation of such acoustic treatments (for special tests at least), though other noise-field study capabilities could become correspondingly impaired and more complex.

6. MODEL-SCALE SIMULATION OF PROPULSION AND POWERED-LIFT NOISE SOURCES

6.1 General Objectives

For clarification of relevant in-flight conditions, selective representation of the primary noise contributions from engine operation is required including engine-airframe interactions, with particular emphasis here on the possible changes in source noise generation and propagation characteristics resulting from the addition of the relative external airflow. Complete aero-acoustic simulation of a practical engine at model-scale is hardly feasible (Fig.10), nor is it necessarily desirable for research aimed towards clarification and evaluation of individual major noise components and possible alleviation. For example, it has already proved both expedient and profitable to simulate separately such specific noise generators of interest as nozzle with jet efflux and fan with intake under forward-speed as well as static conditions. Further engine-components of interest for simulation as 'internal' noise generators include other turbo-machinery (compressors and turbines) and combustion systems, while noise reduction devices and airframe interference also need to be represented. For completeness, it should be appreciated that many of the difficulties now raised in respect of model-scale simulation and relevant rig features apply not only to windtunnel facilities ('fixed'-model), but often even more acutely to mobile facilities, and particularly if equally reliable results are required.

The relative airstream effects to be expected, even for studies of noise from a particular engine-component, are not simple. They can comprise:

(1) Changes in the source noise characteristics arising from the different local airflow and neighbouring surface areas, both internal and external to the engine-nacelle duct.

(2) Modified acoustic near-field development through the local flow field or from local airframe installation interference; including refraction, diffraction, reflection, absorption, scattering and possibly augmentation in the vicinity of the nacelle installation.

(3) Unpredictable development from the acoustic near-field to the aircraft noise far-field, again particularly across practical non-uniform airflow regions or solid surface areas, and with extended sources of a complex nature.

Fortunately, if acoustic and aerodynamic behaviour of the engine-component under static conditions is well understood or can be thoroughly explored, only partial simulation at model-scale may be needed for comparative studies of the primary changes due to forward speed, including the clarification and formulation of basic prediction methods.

6.2 Noise Sources Independent of Tunnel Airstream

Special noise generators whose sound emission characteristics at source are unaffected when placed in an airstream (or change in a known manner) can be profitably applied from at least two aspects, for noise tests in most facilities. Firstly, the validity of conventional or novel noise measurement techniques can be checked when employed within or outside the tunnel airstream, or with a mobile model. Secondly, the particular influence of neighbouring surfaces (e.g. shields) or of flow velocity gradients (e.g. vortex refraction), affecting the near-field and far-field propagation in the relative airstream, can be isolated and diagnosed more readily without simultaneous unknown changes at the source due to the airstream. Some electrodynamic noise sources, (e.g. loudspeakers), jet-resonators (e.g. Hartmann-type), and sirens have already proved useful and are being further developed for such work, particularly with a view to improving their performance in respect of power and frequency range, and to permit controlled variation of their directivity characteristics. However, for acceptable installations in close proximity to surfaces, inside or outside engine nacelles, more compact sources are needed avoiding significant aerodynamic interference on the local airflow.

6.3 Jet Efflux Representation and Quiet Airfeed

Aero-engine jet-efflux development and the associated external jet-mixing noise-source distributions can be investigated at model scale, in principle simply by a geometrically similar jet nozzle, with an appropriate airfeed arrangement providing a quiet air supply to the model (negligible internally generated rig noise) and an acceptable jet-flow profile. For static testing, this now usually presents a straightforward tailoring problem for the particular experimental configuration, involving the incorporation of a silencer, burners or heaters, plenum chamber, and substantial contraction often in close proximity to the nozzle. However, when forward-speed representation is required, such bulky bluff rigs become unacceptable because of their spurious aerodynamic and acoustic effects arising from their interaction with the external airstream. The introduction of conventional aerodynamic fairings to streamline or shield the rig in the airstream can generate its own problems (acoustic, aerodynamic and mechanical), particularly because of the relatively large sizes involved. Such rig problems are naturally tending to become more acute with advances beyond isolated single cold-jet models. Previous experience with jet aerodynamic testing in wind-tunnels is helpful, but alone is completely inadequate for noise-model and airfeed rig design, since good aerodynamic and acoustic simulation is simultaneously required without the introduction of parasitic noise sources. For example, while compactness of the external airfeed arrangement can be achieved in aerodynamic testing by very high pressure airfeeds to the jet nacelle, the controlled expansion (with pressure drop and turning) inside the nacelle to provide a representative flow at the nozzle must now not generate unwanted noise internally, or such excess noise must be controlled by internal absorptive treatment. The difficulties become aggravated with the demand for typical nacelle installations, heated jets, and coaxial or multiple jet arrangements. Relevant practical studies have been started.

6.4 Combustion Simulation Needs

The combustion system, in addition to producing steady-state temperature effects, can generate noise in at least three other ways; directly from the combustion processes, from interaction with the turbine systems downstream, and from interaction with the jet flow. For noise shielding investigations, the first two types (internally-generated noise) may be simulated crudely by incorporating prescribed noise sources within the feed-pipe, for example from internal loudspeakers, a jet hitting a target plate, or even multiple air injectors. But further investigations seem necessary to develop other more suitable devices for installation near to or within model nacelles. The third type, involving essentially the interaction of the unsteady combustion processes with the jet development, probably can be simulated adequately only by producing representative unsteady temperatures in the flow from actual combustion within the model. If this noise generating mechanism is indeed of practical significance, then careful investigations are required to guarantee reliable and controllable simulation of such source characteristics, particularly since external airflow can also affect the characteristics simultaneously.

6.5 Fan Representation and Quiet Drive

Aero-engine ducted-fan representation by small-scale powered-nacelle units generally cannot be expected to offer direct simulation and prediction of full-scale noise levels under forward-speed conditions, in respect of relevant discrete-tones and broadband spectra. For engineering reasons, some important full-scale geometric features such as the number of rotor and stator blades may not easily be duplicated at small scale, the boundary-layer flow characteristics over the duct walls and the blades can be unrepresentative at the low Reynolds numbers, and even the inlet-flow turbulence can differ significantly in intensity and relative length. Nevertheless, such models can be useful at least for diagnostic studies and design guidance, particularly in respect of specific model-configuration changes for which results can be interpreted using theoretical frameworks and thereby applied to estimate the corresponding influence full-scale. The required experimental measurements can then necessitate not only the incorporation of a relatively quiet fan drive, but also the ability to make both acoustic-pressure and aerodynamic-flow studies inside as well as outside the powered nacelle. Separately from noise-source generation considerations, the engine-nacelle flow characteristics and geometrical shape can of course affect the near-field acoustic development in the forward and rear arcs. In principle, for the investigation of such effects, simple high-frequency noise sources of broadband or discrete-tone types can be located within a representative nacelle-duct flow, with the location and directionality characteristics biased as appropriate; naturally, the influence of any variation in duct flow on the noise-source properties must be appreciated. Again, a combination of complementary

experimental and theoretical modelling on particular noise aspects is especially important here for analysis of model-scale results and relevant full-scale interpretation.

6.6 Scaling of Noise Reduction Devices

Noise reduction devices which influence primarily the acoustic propagation towards the measurement point, rather than effecting reduction of sound energy or other changes in characteristics at source, can be subdivided conveniently here into noise absorbers and noise shields. Dissipative-type absorbers whose acoustical performance is determined mainly by viscous flow resistance can often be simply scaled, though the levels of accuracy achievable in the presence of different airflows and at very small scale are not clear, particularly if substantial protective covering has also to be simulated. Resonant-type absorbers currently in use, with perforated sheet facing, are especially subject to significant Reynolds number effects, and it has been suggested that model scaling down below about one-third full-scale requires very careful justification. Indeed, some lack of confidence has been expressed in the practical usefulness of modelling liners in engine ducts at well below full-scale and without detailed engine component representation, for other than basic comparative tests. Shield-type devices usually need to be several wavelengths in size to be effective, so tend to be reasonably large and in principle can be readily modelled if the noise source frequencies are also properly scaled. However, the possible interactions of any aerodynamic flow field with the acoustic field and shield have to be taken into account; in particular the shield boundary conditions should be adequately represented at the shield trailing-edge or the effect of possible variations investigated. Thus, further research on how to model absorption treatment of airframe surface and special shields does seem justified, taking note also of the airframe/engine interference considerations referred to next.

6.7 Airframe Interference Representation

The airframe, apart from providing direct shielding or absorption/reflection properties, can also affect the engine noise characteristics by aerodynamic interactions with the exhaust or inlet flows, and by influencing the acoustic near-field development. Correspondingly, engine airflow in the vicinity of airframe surfaces or edges can introduce excess noise from the airframe (even statically). The external airstream associated with flight conditions may radically modify these effects, while simultaneously generating noise from the airframe which can be significant with landing devices deployed and quiet engine conditions. Here again, the complexities of the related acoustic and aerodynamic effects are so marked that careful selective modelling from both aspects is essential with realistic and well-defined goals. Because of the small amount of experience yet accumulated, any flight research on aircraft noise should invariably be complemented by appropriate model tests, to take full advantage of the possible correlation and clarification of experimental results and the mutual improvement of measuring and analysis techniques. Such complementary experimental programmes have now been undertaken in the UK and USA at least. Moreover, NASA Ames have attempted with some success a few direct tunnel-flight comparisons on small full-scale aircraft, even though handicapped by the high background noise and reverberation effects in the closed test-section of their existing '40 ft x 80 ft' tunnel.

7. CONCLUDING REMARKS

Encouragingly successful noise experiments in subsonic windtunnels have already included basic research studies on single and co-axial jets, jet interaction with airframe surfaces, airframe shielding of engine noise, sound refraction by wing vortex flows, airframe self-noise, engine-fan and helicopter rotor noise. This is not to dispute that, as in the past with aerodynamic and aeroelastic testing, difficulties of model-simulation, experimental measurement and analytical interpretation results will continue to arise with aeroacoustic testing. For example, there have been apparent disagreements and lack of understanding because some forward-speed effects from flight tests on engine exhaust noise and from spinning-rig tests on exhaust nozzle models with internal combustion systems have exhibited a noticeable increase in noise over the *forward* arc, rather than the reduction expected from tunnel tests on simple pure jets. Such discrepancies tend to be aggravated by the individual limitations of the particular testing methods and analytical treatments which can be provided, taking practical account of complexity/cost constraints. Overall, in order to ensure adequate and reliable R&D on aircraft noise under flight conditions, a judicious combination of a wide range of ground-based facilities must still be utilised[1,5], complemented by continual re-evaluation of tractable theoretical frameworks and by carefully-controlled flight research experiments. Nevertheless the critical comments made earlier should be taken to signify realism not pessimism, already implying the attainment of a much greater practical appreciation of viable techniques and of potential improvements than would have been possible a few years ago.

Following on the rapid developments in various acoustic tunnels and the further advances now technically achievable, the provision of noise-models with better selective simulation of engine noise sources (in flight) is next of vital importance. Here the term model is intended in its broadest sense of both experimental and theoretical frameworks, for the complementary interpretation of results for small-scale and full-scale test conditions, from both ground-based facilities and flight. This task presents problems perhaps at least comparable with the complex developments in aircraft aeroelastic models some 40 years ago or in powered-lift aerodynamic models some 20 years ago. Simultaneously, the development and exploitation of directional acoustic receivers and of other discrimination/correlation techniques should also be expedited, to help diagnosis of the changes in noise characteristics with forward speed, and to help isolation of true model-noise propagation characteristics from 'environmental' background-noise interference. An important complementary topic is then the possible aeroacoustic exploitation of modern aerodynamic tunnels, taking full advantage of their good flow

quality and extensive speed range, but without costly acoustic treatment of the existing tunnel circuit and test-section to overcome background noise and reverberation problems.

Finally, I should recall that, under the auspices of the AGARD Fluid Dynamics Panel, informal two-day 'Workshops' involving a small number of specialists have been held on the present subject, in both North America and Europe; the first pair was held during October 1974 (Ref.1) at NASA Langley and at VKI Brussels, the second pair during April/May 1976 at UTRC Hartford (USA) and RAE Farnborough. The resulting stimulating exchange of up-to-date experience, accompanied by debates on controversial issues, proved timely and constructive towards expediting a more integrated and thorough appreciation of relevant technical difficulties and possible solutions. The supplementary Bibliography included here lists about 80 papers issued in 1975/76, which were declared to be of direct technical relevance, though many were not available for reference during the preparation of this report. The principal features and capabilities of appropriate subsonic tunnels have also now been tabulated for circulation primarily to those who contributed (list appended). In view of the still rapidly growing experience in this relatively new field and to help resolve some of the important controversial issues still existing, another International 'Workshop' would be well worthwhile in late 1977.

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LIST OF ATTENDEES AT 'WORKSHOP–76'

At UTRC East Hartford (USA)

1–2 April 1976

Mr R.O.Dietz (AGARD/FDP)
Mr F.C.DeMetz (NSRDC, Carderock)
Dr J.Hardin (NASA Langley)
Mr D.H.Hickey (NASA Ames)
Mr R.Luidens (NASA Lewis)
Dr R.W.Paterson (UTRC, E. Hartford)
Dr R.Amiens (UTRC, E. Hartford)
Mr R.Schlinker (UTRC, E. Hartford)
Mr J.D.Chester (P&W, Hartford)
Mr A.Peracchio (P&W, Hartford)
Mr W.S.Clapper (GEC, Cincinnati)
Dr H.K.Tanna (Lockheed, Georgia)
Dr W.Bhat (Boeing Seattle)
Mr K.J.Young (Boeing Seattle)
Mr F.G.Strout (Boeing Seattle)
Dr I.Ver (BBN, Cambridge)
Mr R.Westley (NAE, Ottawa)
Prof. J.Williams (RAE Farnborough)

At RAE Farnborough (UK)

24–25 May 1976

Prof. M.Perulli (ONERA, Paris)
Mr J.Bongrand (CEPr, Saclay)
Mr J.Hache (Bertin, Plaisir)
Dr F.R.Grosche (DFVLR, Gottingen)
Dr H.Heller (DFVLR, Braunschweig)
Dr W.Bechert (DFVLR, Berlin)
Mr W.B. de Wolf (NLR, Amsterdam)
Prof. J.Sandford (VKI, Brussels)
Mr R.Westley .(NAE, Ottawa)
Mr B.Schofield (HSA, Hatfield)
Mr M.Langley (BAC, Weybridge)
Prof. I.Cheeseman (SU, Southampton)
Mr B.Prichard (SU, Southampton)
Mr M.Cox (NGTE, Pyestock)
Prof. J.Williams (RAE, Farnborough)
Dr T.A.Holbeche (RAE, Farnborough)
Mr T.B.Owen (RAE, Farnborough)

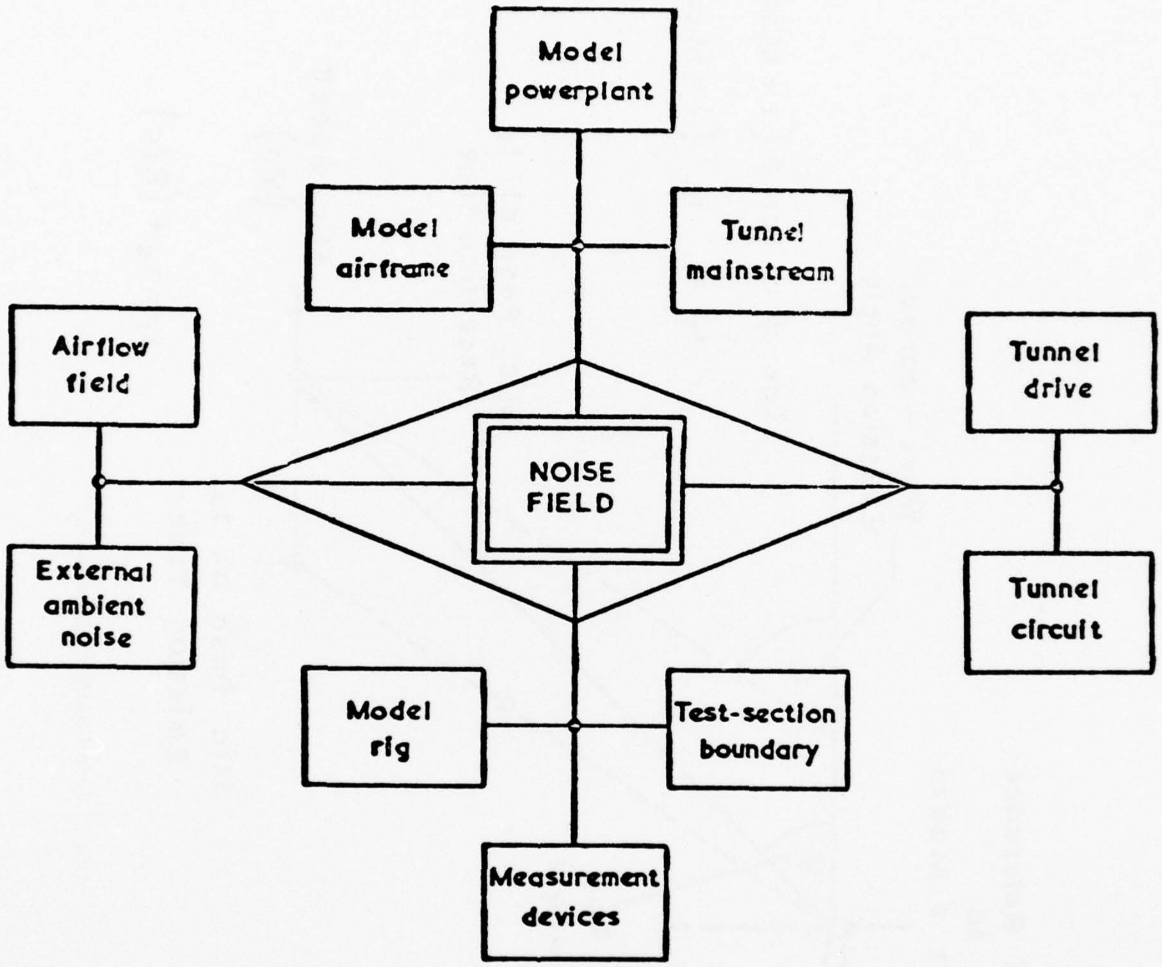


Fig.1 Simplified interaction element diagram

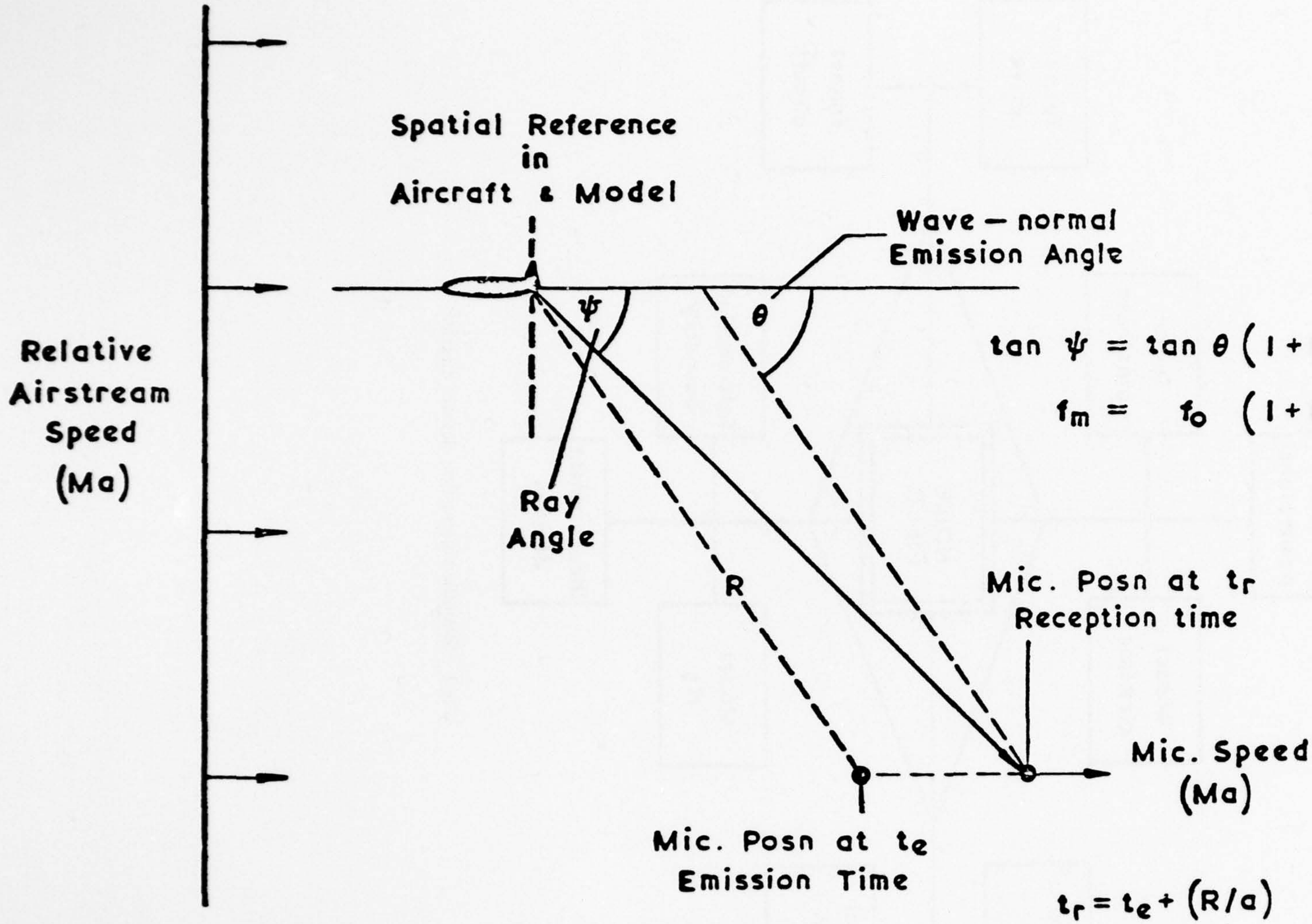


Fig.2 Ideal flight/tunnel equivalence

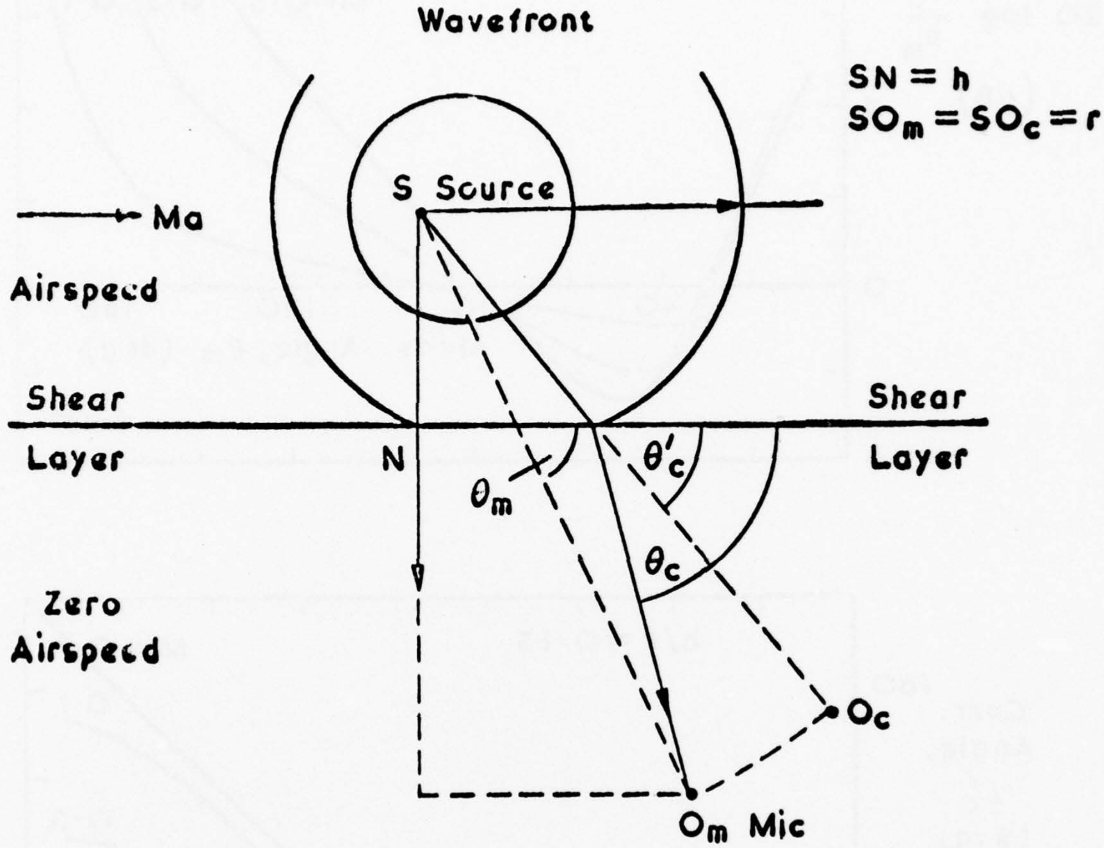


Fig.3 Tunnel shear-layer refraction principle

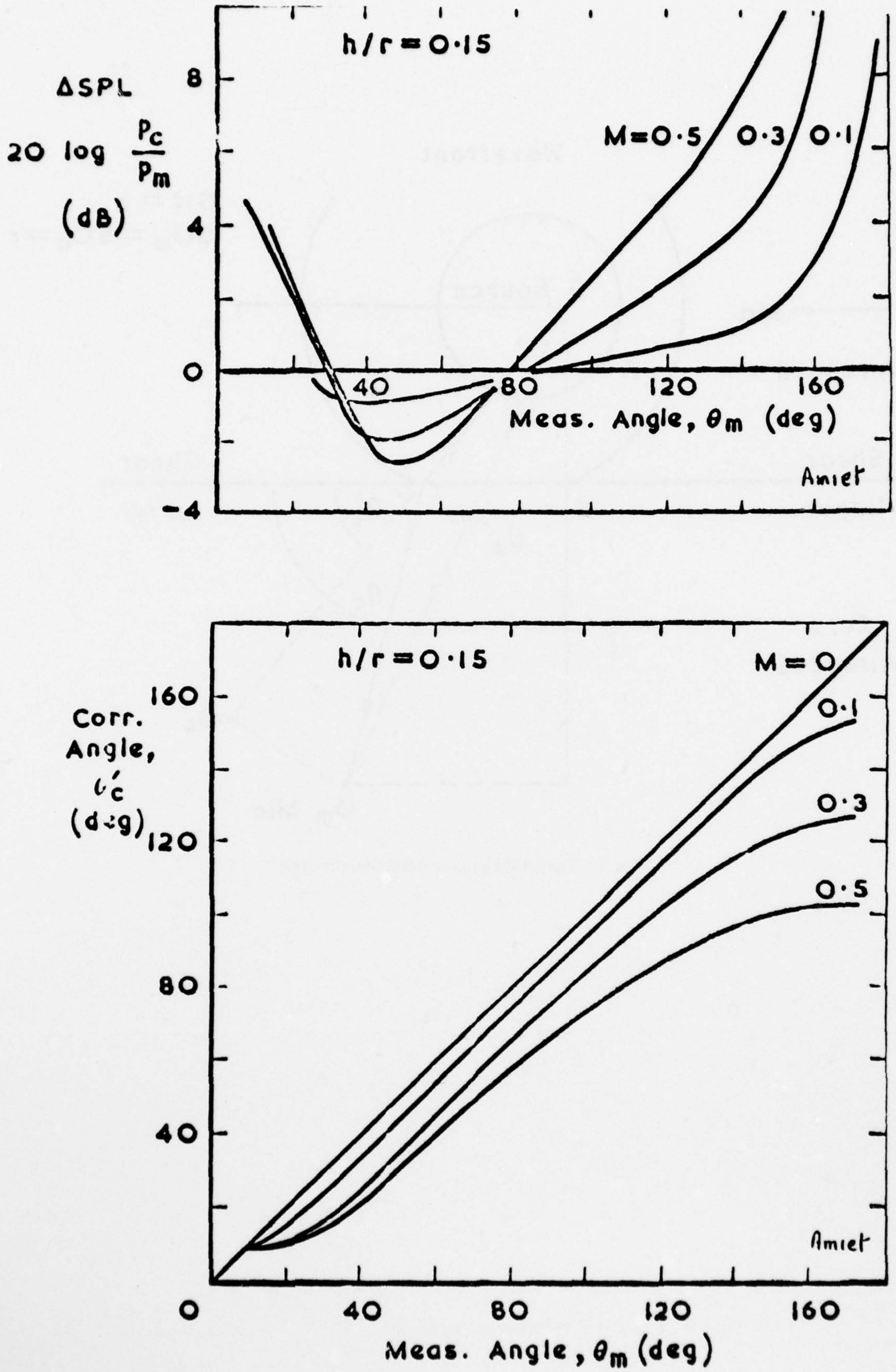


Fig.4 Tunnel shear-layer refraction corrections

For minimum frequency f_{min}

Far-Field in Airstream if

$$R_{air} > R_{far} = k_{far} \times (a/f_{min})$$

$$\text{eg} \quad R_{air} > 1 \cdot 5 \times \frac{340 \text{m/s}}{200 \text{Hz}}$$

$$\approx 2 \cdot 5 \text{m}$$

Free-Field at Mic. if

$$B_{mic} > k_{mic} \times (a/f_{min})$$

$$\text{eg} \quad B_{mic} > 0 \cdot 3 \times \frac{340 \text{m/s}}{200 \text{Hz}}$$

$$\approx 0 \cdot 5 \text{m}$$

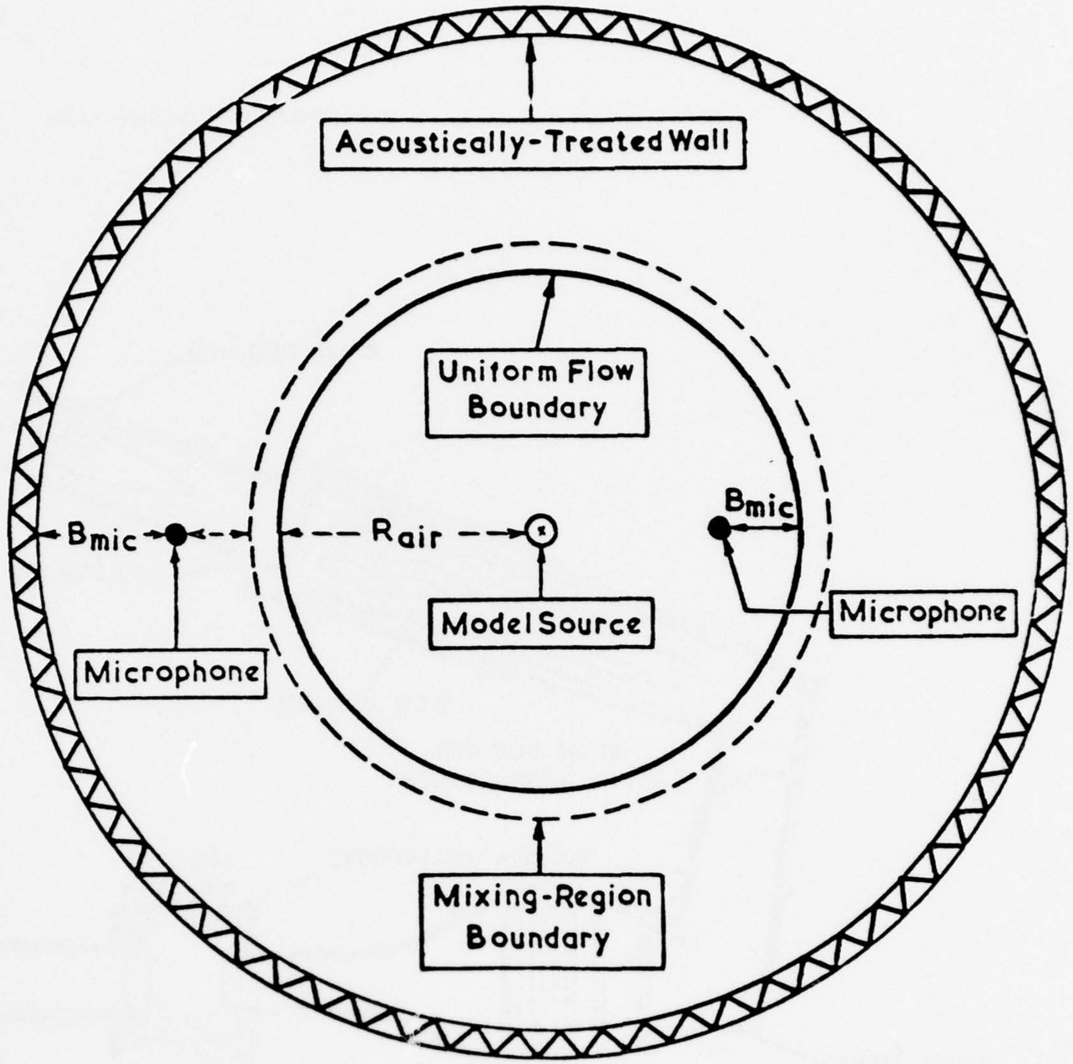


Fig.5 Acoustic constraints on tunnel size

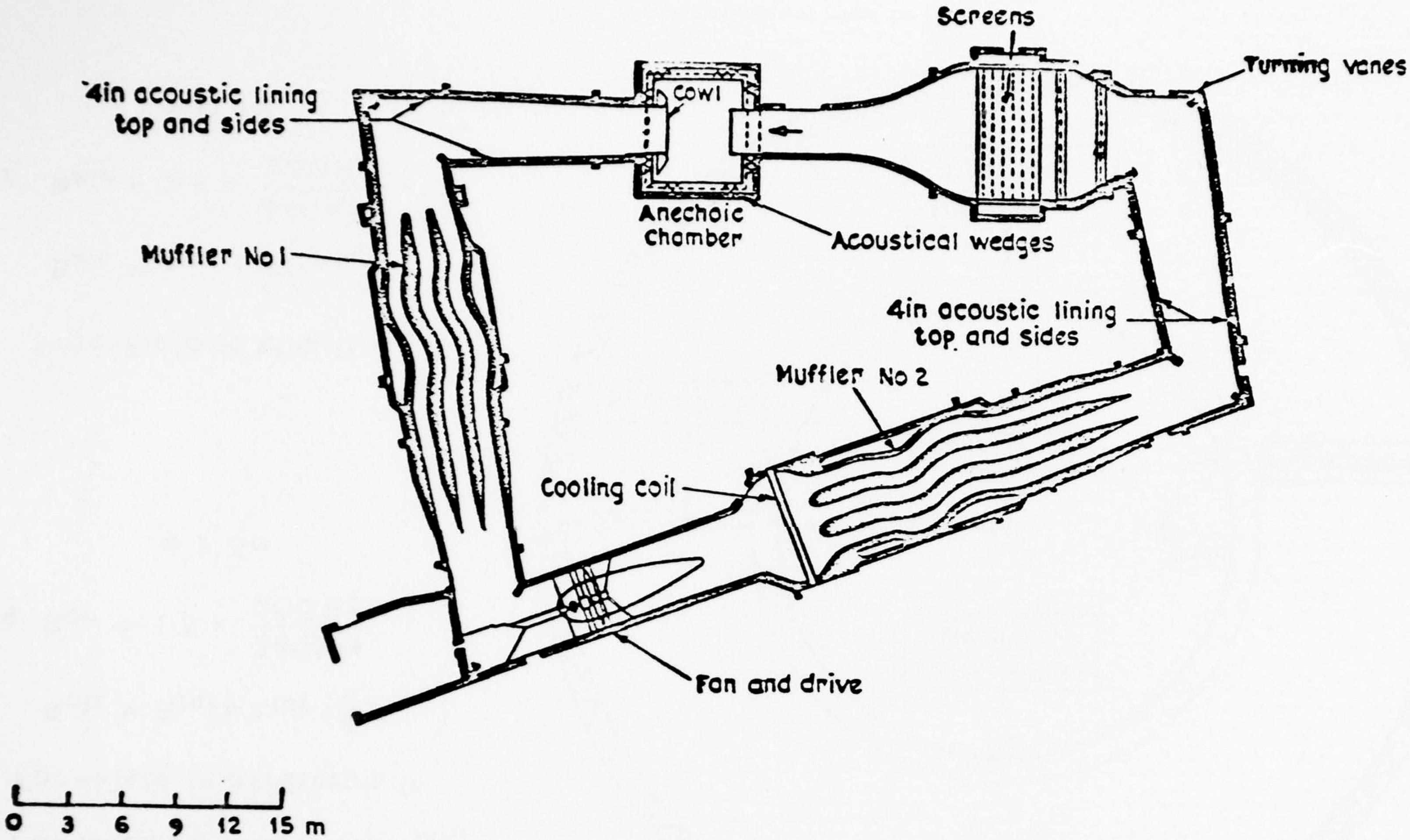


Fig.6 NSRDC anechoic test facility

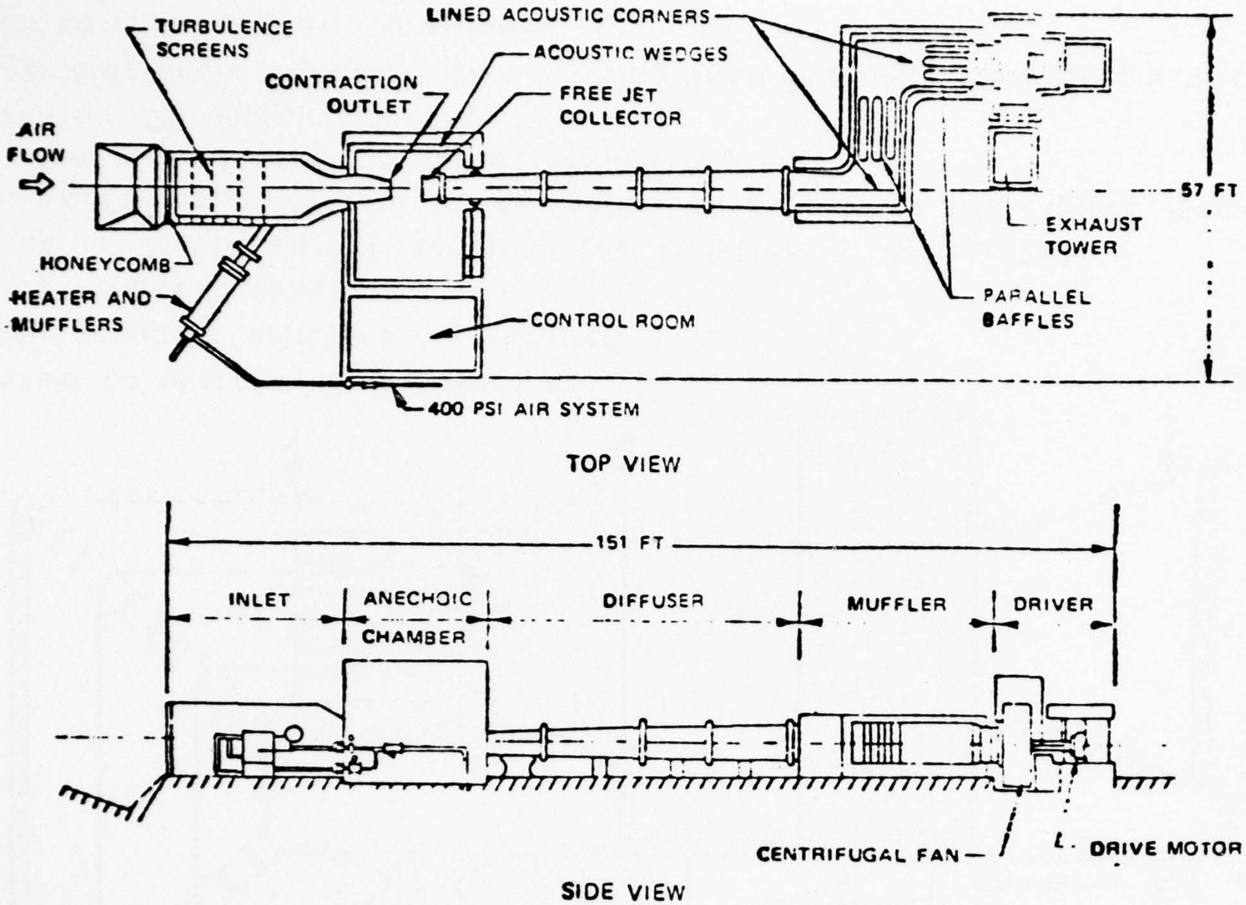
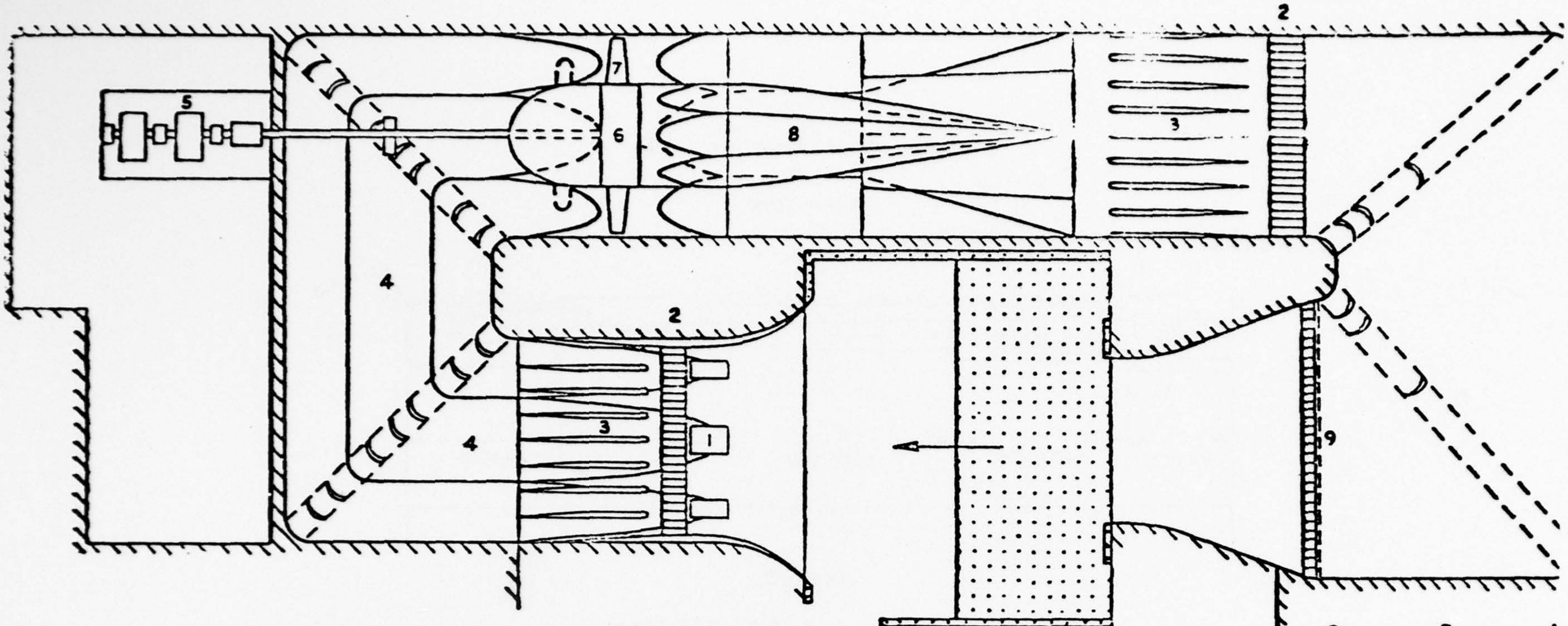


Fig.7 UTRC acoustic research tunnel



1 Collector vented to improve flow steadiness
2 High-frequency acoustic splitters, if required
3 Low-frequency acoustic splitters
4 Square section converted to octagonal by the addition of corner fillets
5 Surplus 3000 kW fan drive system in new motor house replaces life-expired 1500 kW system
6 Tunnel drive-fan repositioned to tunnel return leg
7 New low-noise fan of improved design
8 Multi-passage wide-angle diffuser within existing shell installed downstream of the fan
9 Gauze screen to reduce flow turbulence

Fig.8 Modified 24 ft tunnel retaining 7.3 m diameter nozzle

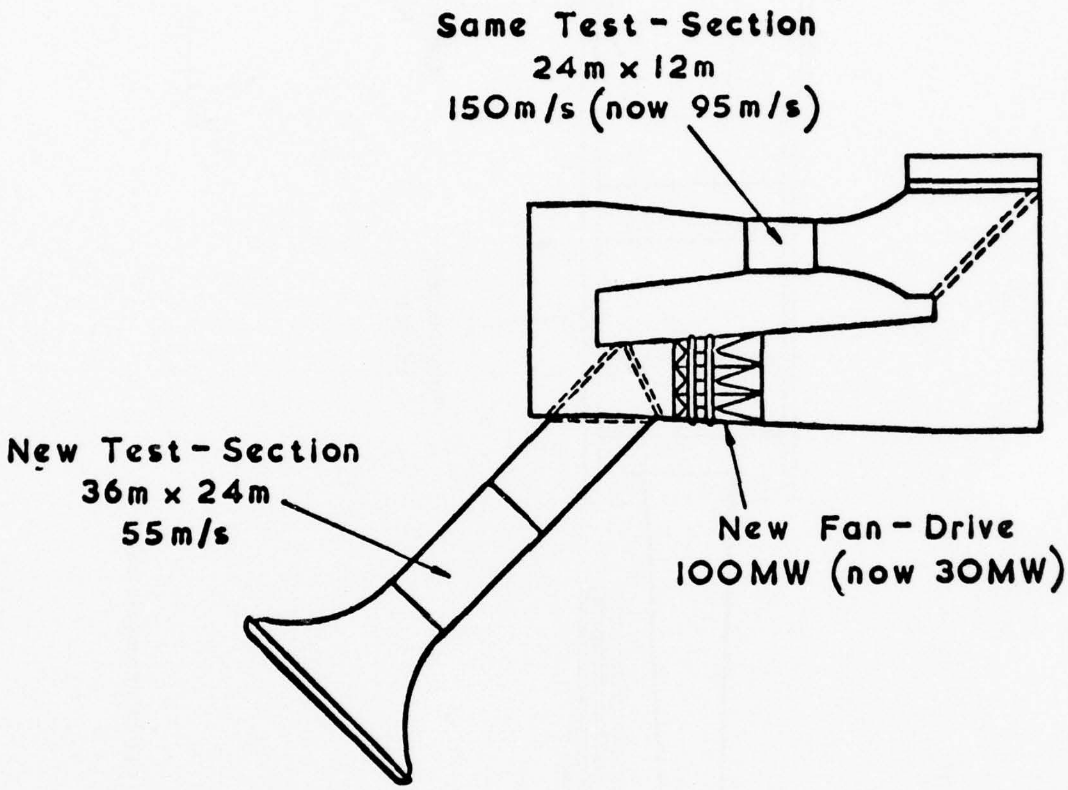


Fig.9 Modifications to NASA 40 ft x 80 ft tunnel

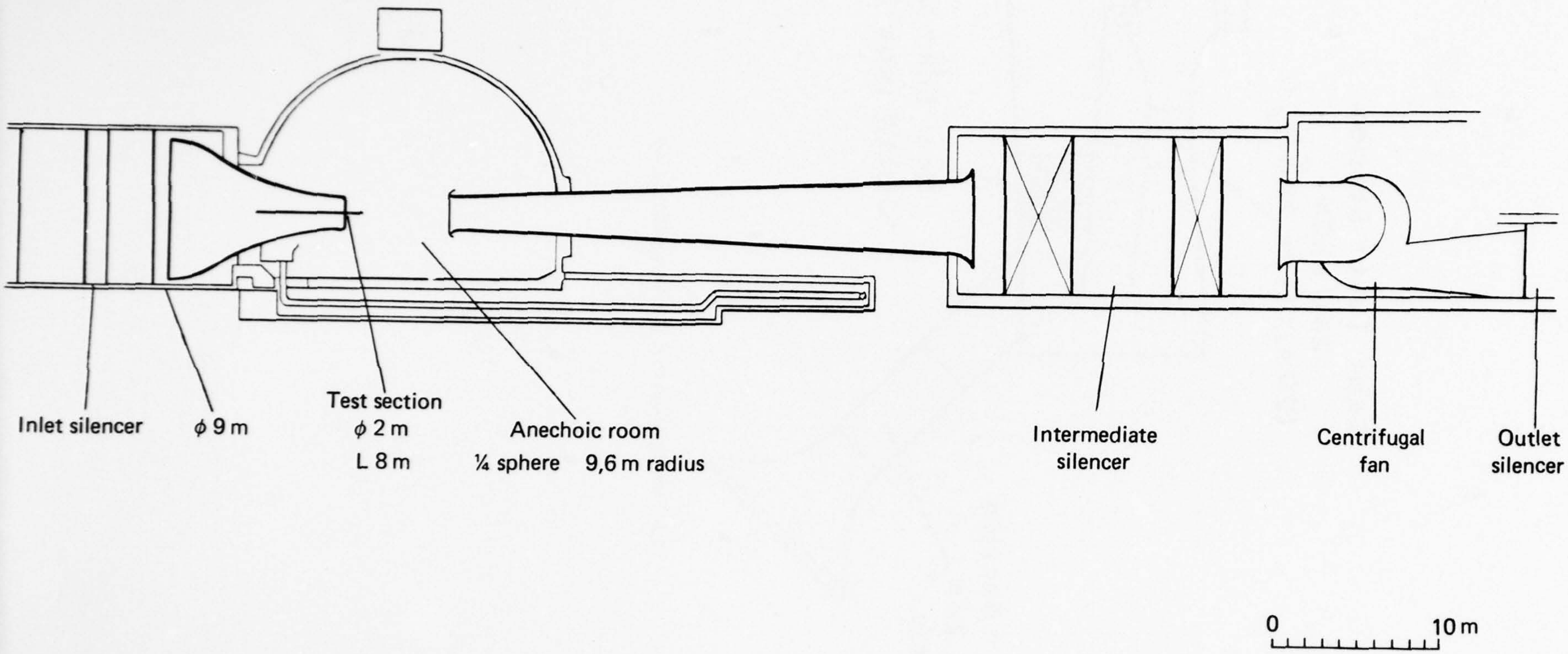


Fig.10 CEPRA 19 acoustic tunnel

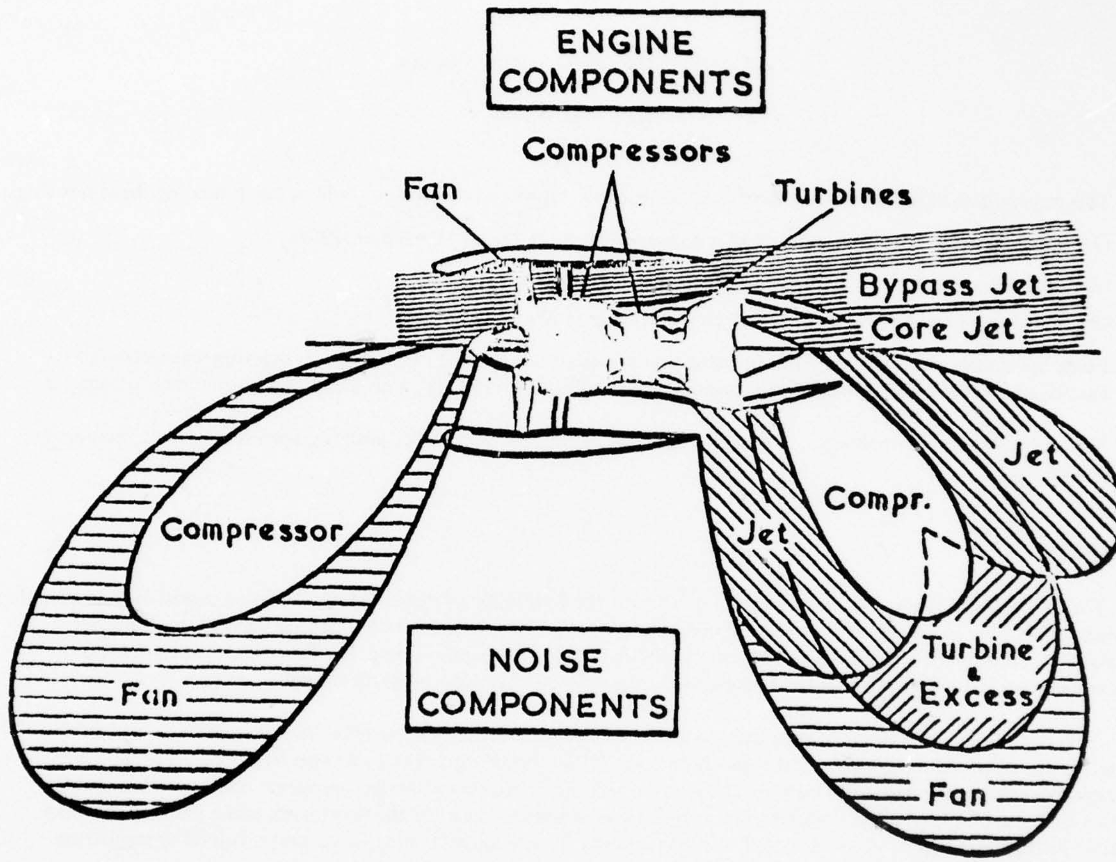


Fig.11 High bypass ratio fan-engine

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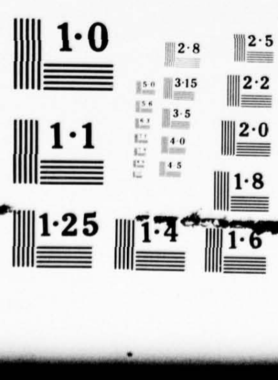

39. Belz, R.A. Bomar, B.W. Goethert, W.H. — *The Development of a Displacement Interferometer for Model Deflection Measurements.* AEDC-TR-76-1976.

APPENDIX 5

MODEL SYSTEMS AND THEIR IMPACT ON THE OPERATION OF PRESSURIZED WINDTUNNELS

by

S.A.Griffin – General Dynamics Convair
M.Brocard – SESSIA/AECMA
M.Bazin – ONERA

This appendix is based upon the discussions of working experts in the subject field at the following three meetings:

(1) West Atlantic experts, held at NASA Langley Research Center, April 8, 9, 1976.

(2) East Atlantic experts, held at AECMA, Paris, France, June 2, 1976.

(3) Joint meeting, NASA Ames Research Center, September 27, 28, 1976.

Forty specialists attended the April meeting, twenty the June meeting, and the joint meeting was restricted to ten. Participants represented government research agencies, private industry, and major educational institutions.

The statements and conclusions in this appendix are, in the opinion of the authors, representative of the participants views.

INTRODUCTION

The objective of these discussions was to determine the feasibility of designing and building model systems capable of withstanding the loads and environmental conditions of High Reynolds Number Tunnels such as the National Transonic Facility (NTF) now in development at NASA Langley Research Center, the Large European High Reynolds Number Tunnel (LEHRT) planned for Europe, and other present day high pressure tunnels.

A review of the discussions held in Europe and the USA reveals that whereas the American NTF (a cryogenic tunnel) is now in development, the European tunnel (LEHRT), based upon the LaWs specification, is still in a configuration development stage, and only recently has it become orientated toward a cryogenic concept. Research relative to the problem of model systems in a cryogenic environment has, for the most part, taken place in the USA, while in Europe efforts have been directed toward planning for low speed testing in a 5 meter tunnel at stagnation pressures of six atmospheres.

With respect to high pressure testing, critical factors include internal balances and support systems, while model instrumentation and materials are considered to be critical to the feasibility of testing in a cryogenic environment.

It is evident that models will be more expensive, and schedules longer if Re approaching full scale are desired. Additional cost is justified, however, by the need for these data in areas sensitive to Re.

Specifically, the meetings were directed toward identification of model system problem areas as follows:

- List primary problem areas.
- Identify existing work in progress and determine if additional effort is required in these areas.
- Determine problem areas where there is no currently planned effort.
- Determine a work-sharing plan for areas with no planned effort.
- Determine an approximate schedule for the above activities.

The conclusions and recommendations are summarized below, followed by a work-sharing plan.

* * * * * * *

SESSION 1. MODEL DESIGN

A. Model Scale and Blockage Criteria[50,51]

1. As model scale is wall design sensitive, it is necessary to refer to the AGARD FDP Transonic Working Section Design Group conclusions.

TABLE 1

Candidate Alloys For High Re Application

Required Characteristics – Strength – Ductility – Fracture Toughness – Corrosion – Resistance – Machinability – Weldability – Stability – Availability

	300K		240K		100K	
	F_{tu}(KSI)	E.10^6 PSI	F_{tu}	E	F_{tu}	E
PRECIPITATION HARDENED						
MATERIALS 17-4PH	210	28	–	–	–	–
PH13-8MO	215	28	220	28	–	–
A286	155	28.5	165	28.5	205	29.5
QUENCHED &						
TEMPERED D6AC	230	29	230	29	–	–
TITANIUM Ti-6A1-4V	150	16	170	16	220	18
MARAGING STEELS						
18 Ni-200	200	27	–	–	–	–
-250	250	27	270	27	330	28
-300	300	27	320	27	–	–

2. A -1% blockage level is presently considered to be standard for most development type aerodynamic models in a ventilated wall transonic tunnel. It is believed that new wall concepts and/or rectangular test sections, will provide for larger scale models, with a significant increase in acceptable tunnel blockage.

3. An increase in acceptable model blockage is worthy of further effort, as the resultant larger scale provides a better opportunity for more complexity and better detail definition.

4. It would be advantageous to include removable walls in future transonic tunnels, to allow incorporation of the latest wall designs.

B. Materials – Developments and Processes[52]

1. Some existing materials offer good potential for cryogenic application.[49] Of these, Maraging 250 Series Steel is perhaps the best present day selection for high strength. Machineability is good in the annealed condition (Rc 33) with age hardening up to approximately Rc 55 possible after machining. (See Table 1).

2. For medium strength high stiffness, A 286 (precipitation hardened) is good over a broad temperature range.

3. Titanium is a potential candidate that is worthy of consideration where model elasticity is a factor.

4. The use of Composite materials in highly loaded models requires further study. Considerable effort is being expended in the use of such materials on advanced airplanes and space vehicles and this work may be beneficial in producing composite materials suitable for use in windtunnel models. Present composite technology indicates that a level of stiffness is obtainable comparable with steel, but that strength is questionable. Strength and stiffness are proportional to the fiber-volume ratio. Its characteristics at cryogenic temperatures need verification, and costs tend to be extremely high.

5. In the case of testing under cryogenic conditions, consideration should be given to maintaining the sting temperature above tunnel ambient by insulation to avoid embrittlement. Insulation would allow use of a broader range of sting materials, and in addition, in the case of a heated balance, would reduce the likelihood of an undesirable temperature gradient across the balance.

6. The data reduction is planned to include a correction for model size, based upon a constant temperature change. Dissimilar materials and corresponding differences in coefficients of expansion, as well as potential temperature gradients through the model will make corrections more difficult.

7. More research and development is needed in the area of use of dissimilar materials at cryogenic temperatures. Manufacturers and the American Bureau of Standards should be consulted in this regard.

C. Candidate High Strength Materials for Use Over Broad Temperature Ranges

1. Many of the available steels offer some of the characteristics required for use in a cryogenic environment.[9]

2. Discretion must be used when selecting a material to assure making a proper choice for the specified task. An acceptable material for cryogenic use may be a poor choice for standard needs, and vice-versa.

3. The values of ultimate tensile strength (UTS) and "E" are lower at room temperature than cryogenic. While the stiffness (E) of a typical candidate is only reduced by 3% to 5%, the reduction in strength (UTS) is 25%. When designing for a selected dynamic pressure in a present day transonic tunnel, (300K) and a cryogenic tunnel (100K), the 300K condition may be more critical.

4. The Maraging Steels (250 Series) offer a good combination of high strength, impact resistance, and fracture toughness.

D. Effect of Model Surface Conditions on Mach Number and Reynolds Number[53,48]

1. Joint mismatches must be avoided. Design discretion will play a large part in alleviating the seriousness of this problem. Critical areas are Wing L.E., Forward Fuselage, Inlet Lips, etc.

2. Tunnel cleanliness is extremely important in order to retain the mandatory high grade surface finish at the model leading edges. A cryogenic tunnel such as the NTF should provide the desired cleanliness as the planned venting of approximately 1% of mass flow will tend to remove much of the tunnel contamination.

3. The use of LN_2 indirectly curtails the inducement of foreign matter into the test section.[31]

4. The development of an acceptable model surface filler material for high pressure tunnels is extremely important, particularly in the case of a cryogenic tunnel. Such a filler material must possess the characteristics of tenacious adhesion, quick removal, and a smooth hard finish. Application should be simple and cure time short. These characteristics must be maintained over a broad temperature range. The importance of the filler material should not be underestimated, and an experimental program to verify the required characteristics is justified.

5. Polishing to a shiny finish does not necessarily assure a high quality finish.

6. Admissible roughness estimates (surface finish), are directly related to model scale and desired Re. Studies (Ref. 2) show that 16 micro-inches at the Wing Leading Edge would be acceptable in most cases. Experimental studies are needed to verify this.

7. The location and size of pressure tube orifices requires special consideration, and work in this area is presently planned (Ref. 36). In the case of a chordwise row of orifices, the desired close spacing at the leading edge is a potential source of error and must be treated with caution. The present practice in 2 dimensional high Reynolds number tunnels of staggering the pressure orifices is a potential solution, providing that corrections can be made for the difference in spanwise locations.

8. Special emphasis should be placed on some basic design rules when designing a model for a high Reynolds number test. Careful attention should be given to the design of wing/fuselage attachment joints. Streamwise parting lines (located in the least critical flow area possible) should be used. It should be noted that the deflection between two parts in a joint under load is much more critical due to the added emphasis on surface finish. Basic split lines and attachments will greatly influence the ultimate surface finish achieved on the model. Tradeoff studies will be needed to obtain the maximum versatility/structural integrity/surface finish required for each test situation. This whole area is worthy of experimental study.

9. Pressure to meet design criteria will lead to the evolution of new techniques and methods in model design and fabrication.

10. When considering composite materials for use in a cryogenic environment there was some concern about a possible crystallization of the local surface, and its adverse effect on the surface finish.

E. Model Complexity, Cost, and Schedule[54,55]

1. The degree of model complexity that can be achieved is directly related to configuration, model scale, and desired Re (model load). The cryogenic tunnel offers a better opportunity to achieve a higher Re, providing special attention is given in the design to the environmental conditions.

2. The need for testing development models at higher Reynolds numbers was expressed.[15] Such models usually require variations in leading edge shapes, trailing edge shapes, inlet lips, etc. These variables are often in areas where surface finish is critical. The alternative to the multi-piece model is of course other complete models, or for example, alternate wings. Cost and schedule is impacted both in the model and in testing, and in a similar manner to today, such costs will be reviewed during the initial stages of a model program. Schedule improvement can be achieved by an early test of the model with less variables; for example, a less sophisticated wing.

3. The matching of inlets/exits, and the need for internal flow ducting within a fuselage greatly increases the model design problem. It was generally felt, however, that they are required.

4. Costs will increase if the flight Re for a given configuration is desired. Design costs may be 100% greater than present day.[48]

5. Fabrication costs may increase by approximately 30%, with overall model costs increasing by approximately 50%.[2] There was some concern that costs might be higher and possibly preclude a high Reynolds number test. Generally speaking, however, it was felt that such testing was highly desirable but that it would be limited to conditions and configurations that were felt to be Re sensitive.

6. Schedules will be longer, primarily because of the need for additional analysis, (i.e., stress, deformation prediction, etc.), prior to the start of fabrication. In the case of development models, there was some feeling that the longer schedule could be more critical than the increased cost.

7. Quality control and inspection of models will require a greater effort.

F. Design of Models for Cryogenic Environment

1. Model testing over a broad temperature range will result in model dimensional changes. Such changes, if not taken into consideration, may directly affect the accuracy of test data. For example, internal flow measurement is dependent upon the accuracy of the internal duct geometry.

2. Model design specification will be highly definitized. It is acknowledged that for a given dynamic pressure, Re can be increased by a factor of 5, simply by reducing temperatures from 300K to 100K. This significant advantage can be achieved by designing a model for use in a present day transonic tunnel, comparable in size with, for example, a cryogenic tunnel such as the NTF. The model design criteria must be based upon the selected dynamic pressure of the present day tunnel and the cryogenic environment, which will influence selection of materials.

3. Experiences gained in the first development models for NTF will undoubtedly offer a reduction in engineering costs for subsequent models.

4. A development model designed for NTF is in the planning stages.[6] It will serve to evaluate possible fabrication problems and techniques in working with hi-strength steels. Initially it will be tested in the NASA 8-foot pressure tunnel.

5. It is generally felt that a cryogenic environment precludes the use of dissimilar materials. If true, this would have a definite effect on the model cost factor. Further study of a more detailed nature is required, and an experimental study is justified. It would be particularly advantageous to be able to manufacture the difficult low-load carrying areas such as the duct of more easily workable materials.

6. A definition of baseline materials, procedures, use of dissimilar materials including fasteners, fillers, potential use of composite materials, etc., must be gathered together as the basis of a users handbook; contributions should be sought from all sources.

G. Simulation of Inlet/Exit Conditions (Engine Simulators)[56,62]

1. Provision of inlet and exhaust simulation in those test programs where transonic flight conditions are a major design consideration, is highly desirable. This will require internal flow, with attendant instrumentation.

2. The use of engine simulators or ejectors[42,60] is mandatory for simultaneous matching of inlet/exit conditions. Further investigation is needed to verify the feasibility of operating simulators in highly pressurized windtunnels. The cryogenic environment presently contemplated for some high Reynolds number tunnels is another area of concern in the use of simulators. It was generally felt, however, that while the problem is a real one in future cryogenic tunnels, it should be considered as secondary to more pressing problems.

3. The above models will definitely fit the category of being complex. Simulation of mass flow, inlet/exit pressure ratios, etc., will require design and fabrication techniques employing the use of multi-piece construction, possibly dissimilar materials, and constraints which tend to complicate the end product.[11]

4. Exit plane and internal duct drag rake instrumentation should be of concern as regards structural limitations due to air loads.

5. Proximity of balance to flow-through ducts and/or engine simulators provide thermal flow paths which will induce thermal gradients across the balance, directly affecting accuracy.

6. The need for on-board compressed gas for simulator/ejector operation will necessitate use of an air balance of small diameter (2 to 3 inches). The thin walled bellows used in such devices will require special attention if used at low temperatures. Changes in spring constant will adversely affect accuracy.

H. Designing for Low Safety Factors

1. *Standard practice in present day fan-driven tunnels is a safety factor (SF) of 5 on ultimate or 3 on yield,* whichever is greater. Some exceptions are taken when proof loading is accomplished. If a SF of 5 is

mandatory in the new high pressure tunnels, the allowable working stress of acceptable materials will severely limit our ability to achieve Re approaching full scale.[43] Non-return type tunnels will accept a lower safety factor, usually 3 on ultimate, and in France in some cases an acceptable safety factor is 1.33 elastic limit and twice the breaking limit.

2. Today's SF is required for protection of the tunnel fan, and can be attributed to questionable load prediction, including dynamic effects, insufficient quality control/inspection, etc. A lower safety factor, while desirable, can only be justified by much greater effort during design and construction. In addition, critical model components must include the means of monitoring loads during a test. (Ref. Session 4 H1).

3. Improved and more in-depth model engineering will, it is felt, make lower SF's more acceptable to the facility.

4. Devices are employed today within models and/or facilities, to rapidly reduce dynamic pressure or change model attitude to unload the model.

5. The achievable degree of complexity in the model, (multi-piece) is directly a function of design allowables.

6. To provide a better simulation of airplane flight geometry, it may be advantageous to induce wing deformation by increasing stress, (i.e., reducing SF). While desirable, the established SF design criteria must be met.

7. The question of acceptable safety factors needs to be resolved with the facility engineers; model failure is unacceptable. Safety devices such as screens, or "Q Reducers" should be considered, but the resultant power loss is undesirable.

J. Quality Control and Inspection

1. A much greater effort is anticipated in this area, from raw material procurement to final assembly. Certification will be required as proof of satisfactorily meeting material specifications and processes.

2. In certain cases, proof loading will be accomplished and documented.

3. Inspection data will be recorded to demonstrate that model meets design criteria and safety standards.

4. In general, the models designed for high Re will be governed by similar constraints as an airplane with respect to safety factors, operating envelope, and quality control. The obvious impact on model cost and schedule is considered to be a necessary and acceptable penalty to pay for the added benefits of high Re testing.

K. Design of Aeroelastic Models for Testing at High Dynamic Pressures[63,64]

1. No position has been established in the USA at this time as to the need for testing true aeroelastic models in NTF. In Europe, however, it is felt that the ability of the Researcher to provide dynamic similarity of flutter models in a tunnel operated at room temperature is limited both in Mach number and Reynolds number scaling. These limitations can, it is believed, be partially overcome as regards the Mach/altitude flight envelope, by testing at low temperatures with a possible stagnation pressure of 6 bars.

* * * * * * *

SESSION 2. MODEL DEFORMATION

A. Measurement of Model Deformation and Attitude in the Tunnel[65,66]

1. Model deformation will occur particularly at high Re, and there is a need to measure it.[49] The ultimate deformation measuring system (DMS) has not been identified. Work is continuing on various systems at NASA-LRC,[7] AEDC,[40] (US), and in Europe. It is generally felt that this is an area of extreme importance justifying research on a broad front. The recent work of General Dynamics/Fort Worth with the stereophotographic system[77] and its use at NASA-Ames and NASA-Langley was recognized. Such a device has also been used with success in the ONERA-MODANE test center, to identify the bending and torsion of helicopter blades on a 4 meter rotor.

2. A stereophotographic DMS has been successfully used in transonic tunnels at NASA-LRC, (8' TPT), and NASA-Ames (11'). The use of this system, however, in a cryogenic environment, or for that matter, any DMS presently under review is questionable, and deserves further attention..

3. The need for on-line data is desirable. Tunnel operating costs, however, would become prohibitive if analysis of on-line data were required before proceeding to the next test point.

4. Deformation data available within 15 to 30 minutes may be acceptable for the moment. Real on-line data will be necessary in the future.

5. The DMS will provide improved analysis and interpretation of test data by direct measurement of:

   - Model attitude;
   - Control surface deflection;
   - Aeroelastic deformation;
   - Direct sideslip measurement;
   - Store separation characteristics.

6. Some DMS systems will require the installation of reference reflective inserts in the model surface.

B. Model Instability Due to Large Deformation

1. Failure of model components represents a potential problem; facilities may require users to install on-board "rapid response" instrumentation devices (i.e., accelerometers), to indicate impending instability of critical components such as wings, tails, etc. This practice is relatively common in present day facilities (Ref. H1).

C. Matching Model/Airplane Deformation in Tunnel

1. Future test requirements have indicated the desirability for a better match of model/airplane deformation. The DMS will provide the tool for comparison of predicted and achieved model deformation.

D. "Tuning" Model Deformation by Variation of Tunnel Temperature and Dynamic Pressure[18]

1. Testing in a variable density/temperature facility provides the capability of independent control of Reynolds number and dynamic pressure at constant Mach number in the aeroelastic mode of operation. Wing loading can be varied while maintaining constant Re, providing the capability of tailoring a model wing shape to more closely match the desired shape. Allowable wing stresses must, of course, be monitored.

   NOTE: Deformation of models in a high pressure facility is a certainty. A means of accounting for model elasticity is necessary before the test data can be rationally applied to the full-scale design problem. Simultaneous solution of equations which contain both the aerodynamic influence function and the structural influence function are necessary for this task. Methods do exist for the prediction of load distributions on an elastic airplane wing, and the corresponding deformations due to twisting and bending. A program based upon a modification of the Weisinger L-Method is one example presently in use.

* * * * * * *

SESSION 3. SUPPORT SYSTEMS

A. Conventional Support Systems – Sting/Model Interference – Divergence[67]

1. Selection of an optimum support system to achieve the designated test objectives, remains as a critical decision, subject to considerable compromise. Conventional support systems such as stings, blades, struts, and wing-tip supports, will require evaluation, with a selection based upon test objectives, and an engineering opinion of least, or perhaps known values of interference. This present day problem is accentuated in the high-pressure facility, by the need for larger support systems. Uncertainty in applying corrections may limit the size of the support and prevent the user from utilizing the full capability of the facility. A better understanding of support system corrections is mandatory.

2. Sting allowable stresses must remain comparatively low to avoid divergence. Each joint in the support system is an additional problem, and "one-piece" sting/balances should be considered. For certain configurations, multiple stings are advantageous.

3. Information presented by NASA-LRC indicated a need for a sting diameter approaching 5" for a maximum Re case (9 bar) in the NTF. This is predicated upon the need for a constant sting diameter aft of the model base. Such a sting would severely compromise the empennage of most configurations.

4. New materials are needed with greater allowable stress/stiffness characteristics. Composite materials are a possibility.

5. In the case of a cryogenic high pressure facility, the low temperature does improve the properties of selected high strength materials without adversely affecting ductility and fracture toughness.

6. Sting adapters will be necessary to allow smaller stings in the lower load cases.

7. Aerodynamic data is severely compromised by sting interference and a solution must be found.[19] Sting supported models designed for high loads will be basically sting/balance limited.[11] With most sting support models, internal flow passages, the balance cavity, sting-to-model clearances, and model geometry combine to

limit the allowable size of the sting and balance. Acceptable distortions of the aircraft geometry are configuration oriented, and must be weighed against the need for high Re.

B. Dynamic Behavior of Model/Sting/Balance

1. For each support system, sufficient analyses must be performed to establish:[37]

   – Dynamic characteristics,
   – Stability characteristics, static and dynamic,
   – Induced angle of attack effects,
   – Structural integrity.

C. The Magnetic Suspension System (MSS) and Its Influence on the Model System[68]

1. The successful development and operation of a superconductor magnetic suspension system has created new interest in the feasibility of such systems, particularly in view of its compatibility with a cryogenic tunnel. The consensus of opinion, however, was that the MSS, while highly promising and desirable, does not offer a new-term solution to the support system problem, and that in relation to other areas requiring further study, it should be considered as lower priority. It was recognized, however, that a successful MSS does offer an ideal solution to the support system problem, and that a continuation of research effort is justified.

2. Feasibility of MSS for high Re facilities has been established on the basis of:[12]

   – small-scale prototype demonstration with superconductor system,
   – scaling calculations verified with small size coils,
   – assumed compatibility of MSS with cryogenic windtunnel operation.

3. Outstanding problem areas where additional research work is required:

   – shape-independent model position sensors,
   – detailed design of large aerodynamic models,
   – aerodynamic measurements,
   – model launch techniques,
   – reliability of MSS support.

4. Logical next steps:

   – implement MSS for LRC cryogenic pilot tunnel facility,
   – advance state of the art in outstanding problem areas as much as possible.

5. Use of the Electromagnetic Position Sensing (EPS) system seems to be compatible with the use of the Deformative Measuring Systems (DMS) that were described.[28] The EPS operates at 20 Kilohertz in a narrow band. For extreme accuracy in determination of angle of attack (±.015 degree) we use a laser scheme. In this case the laser could serve as a reference for the DMS, or perhaps the DMS could also provide position signals for the MBSS. In either case, no basic incompatibility exists.

6. The question of the model design itself and the model injection systems contain the most unresolved issues. We have solved these problems at the existing MBSS size. At this point in time, we are leaning toward elliptic magnetic cores and non-magnetic external contours. Again, we are leaning toward composites but feel many of these matters should be explored on the 15 inch scale.

   Finally, magnetic forces and moments are proportional to the volume of magnetic material while aerodynamic forces are proportional to area; ratio of volume to area increases with linearity, with increases in size. Thus, by increasing tunnel size from 15 cm to 50 cm tunnel, the volume surface ratio increases 10/3. In other words, the problem of generating magnetic forces is simplified with larger size models. From this standpoint of increased Reynolds number which implies an increase in dynamic pressure, (which results in the need for increased magnetic volume), is partially compensated for by retaining the amount of magnetic material in some proportion as is common now. Since we operate a long way from saturation, I would guess the scale up is possible. Again, it would be nice to do the intermediate step first.

7. The influence of the MSS on the model system is cause for concern, and further investigation is needed in the following areas:

   – use of magnetic material in model,
   – on-board instrumentation,
   – size of magnetic core in model,
   – how data will be recorded from on-board instrumentation (must it be telemetered?)

* * * * * * *

SESSION 4. INSTRUMENTATION AND BALANCES

A. High Capacity Balances

1. Current research toward higher capacity/improved accuracy/cryogenic compatibility, and new concepts, was recognized as taking place in Europe[74] and the USA.[8] It was generally felt, however, that this was an area requiring special effort on a broad front, and that coordination of the working engineers should take place in the form of "workshops", to facilitate an exchange of information.

2. Beam (one piece), two-shell (floating frame), and dynamometer assembly balances,[71] (with or without flow-through), all have potential for use in the high pressure tunnel. While the cryogenic tunnel does result in a lower dynamic pressure (q) for a given Re, loads do remain high compared with today's transonic tunnels, and high capacity in relation to diameter remains an important issue.

3. Model sizes for a 2½ meter tunnel (NTF) will be similar to those for Calspan. This size of model would typically use a 2½" diameter balance for high load cases indicating the need for high capacity. High capacity must be maintained in a combined load sense; for example, pitching moment and normal force must be high to offset the need for moving a balance within the model during a test. Balance stiffness is critical to alleviate divergence.

4. Use of a 5 component primary balance with the drag measured at the sting, or by pressures, offers additional stiffness, and the potential for increased capacity.

5. The two-shell is a versatile design because of the relatively large hole through the balance center,[17] which allows passage of "on-board" services, without a significant decrease in capacity. This concept was shown to be particularly suitable as an air balance.[4,5] An earlier discussion indicated the need for matching internal flow, and the use of engine simulators. For this type of testing, an air balance is mandatory, and its use in a cryogenic environment needs investigation.

6. The variable range concept was discussed. While the purpose was recognized, the consensus of opinion was that basic balance problems must first be resolved.

7. A standard definition for balance capacity is needed. It should be established to include combined loads.[17]

8. Individual elements for each balance component provides a better stress distribution and a reduced thermal gradient. Increased capacity relative to size is, however, questionable and needs further study.

9. Balance capacity to size ratio will limit the ability to use higher pressures in a windtunnel and restrict the simulation of the flight envelope for both fighter and commercial aircraft. Increasing Re by an increase in test section pressure only, is therefore limited.

B. Environmental Control of Balances (100K to 300K)

1. The need for environmental control of the balance is not finally determined. While it was generally agreed that no control simplified the installation in the model, the majority of specialists felt apprehensive about achievable accuracy without control.

2. Research is underway to determine the feasibility of allowing the balance to function at tunnel temperature,[8] thus alleviating the need for inducing and controlling heat to the balance system. Accuracy was questioned, particularly the ability to temperature compensate over a wide range of temperature.

3. With changes in tunnel temperature, the model is likely to follow the tunnel much more quickly than the balance, creating an unacceptable temperature gradient. The time required to allow the model/balance system to stabilize will be very expensive.

4. The 1/3 meter cryogenic tunnel of NASA Langley Research Center is an example of an excellent test facility for the resolution of the aforementioned balance problems.[35] Research is recommended using a balance size compatible with use in future 2½ meter tunnels. A test of a 2½ inch diameter balance covered with a simple cylindrical body shape was suggested as a research test for the 1/3 meter tunnel. It was felt that the larger balance would provide more meaningful data on temperature gradients.

5. Temperature compensation should be preformed incrementally for the full temperature range of each tunnel, (+155°F to −300°F). The increments are important, since thermally induced microstrain curves are non-linear.

6. A temperature gradient is likely to occur with or without environmental control and is a function of the model/balance, and sting/balance attachment. This again justifies research on a number of balance concepts where attachments vary, and where different methods of heating can be employed. For example, temperature, control of a beam balance revealed that it was difficult to heat the balance uniformly because the balance to model taper remained colder. The two-shell or floating-frame concept with its center hole offers the alternative of internal heating through the length of the balance, and a different model to balance attachment. Model/

balance and balance/sting attachments modified for inclusion of thermal insulation to reduce temperature gradients, may be weakened unacceptably.

7. It was generally agreed that the balance and its accuracy is a very key unit in the model system, and that research should be pursued in all areas; heated/non-heated balances and various concepts. The use of cold flow (cryogenic) pipes should be considered as a means of low cost research.

C. Balance Accuracy – Contributing Factors

1. Accuracy requirements for future high pressure tunnels will be at least as good as those specified for present day transonic tunnels. This would be 1/2 of 1% of maximum, under combined loads. A requirement of 1/10 of 1% was demanded for one European transport program.

2. Contributing factors resulting in degradation of accuracy:[16]

   - Temperature gradients across balance.
   - Operation of a balance at less than full scale loads.
   - Insufficient calibration.
   - Inadequate calibration equipment.
   - High stresses.
   - Calibration under cryogenic conditions will be needed to determine effectivity of the heating system.
   - Operation at high pressures – effect on strain gages.
   - Temperature compensation, if heat jacket is not used.

   Most of the factors listed above are present-day problems that are likely to be accentuated in the future high pressure tunnels. A better understanding of them is part of the additional effort required in engineering that results in the aforementioned increased costs and schedules.

D. Fatigue – Use of Karma Gages

1. The Karma gage has a significantly better fatigue life at higher strain levels. This substantial gain becomes very important when fatigue is used as a criteria for allowable gage stress. The Karma gage allows the designer to make better use of high strength steels in the design of high capacity balances.

2. Karma gages can be compensated for modulus and zero shifts.

3. The cost of compensation and installation of Karma gages was thought to be higher than the standard Constantan gages; however, NASA-Langley is now using it as their standard gage on all balances.[8]

E. The Magnetic Balance and Suspension System (MBSS) as a Force Measuring Device

1. Actual experiments at M.I.T. give confidence that the accuracy of MBSS as a force balance is acceptable and practical. The accuracy is at least equivalent to sting systems, in small scale facilities.

F. Instrumentation – Miniaturized Instrumentation

1. As indicated earlier, there is a need for development testing in high pressure cryogenic tunnels. Models for such tunnels will not be large, and space will be at a premium. Miniaturized instrumentation with no loss in quality is highly desirable. An example of effort in this direction is the work in progress at NASA-Langley on a multiport electronically-scanned pressure sensor (MESPS), capable of operating over a wide temperature range.[7] Progress is also being made by transducer and scanivalve companies.

2. The use of actuators within a development model is a common occurrence, particularly in the case of high cost facilities where entry into the tunnel for model changes is expensive. Compatibility of actuators and bearings with a cryogenic environment, however, is cause for concern, and justifies investigation. Hydraulic fluids that maintain their properties under high pressure and at cryogenic temperatures are available.

G. Flow Visualization

1. A need for this was expressed. The 1/3 meter cryogenic tunnel offers an excellent tool for a study of various techniques. Some of these techniques do require modifications to the model.

H. Model Surface Instrumentation in a Cryogenic Tunnel

1. Previous discussions indicated the need for various types of surface instrumentation in the model. For example, wing bending moment strain gages, accelerometers for buffet, and local pressure transducers. It will be very difficult to thermally protect such instrumentation, and their operational characteristics over a broad temperature range needs further study.

2. The need for local balances for measuring/isolating the forces on individual components was indicated. Such balances and/or root bending moment bridges were suggested as a means of monitoring critical component loads, thereby allowing a reduction in safety factor. Again, the environmental impact on accuracy needs to be addressed because thermal control is not feasible for this type of instrumentation.

J. Effect of MBSS on Model Instrumentation

1. The compatibility of MBSS and model instrumentation is of very real concern. On a near-term basis, models will be sting supported in the new high pressure facilities; however, the MBSS does have significant advantages, and its future use will certainly be curtailed if there is an adverse effect on model instrumentation.

* * * * * * *

SESSION 5. TECHNIQUES AND SPECIALIZED EQUIPMENT

A. Use of a DMS in the New High Pressure Facilities[45]

1. There is general agreement that a DMS is necessary in the new facilities, and while the ultimate system has not been determined, consideration must be given to providing for its installation. Each potential system must be reviewed in relation to difficulty of installation and its compatibility with the cryogenic environment.

B. Model Handling Techniques in a Cryogenic Tunnel[6]

1. Present plans are to pre-assemble the model and support system in a preparation area. Checkout of all systems will occur in that area, with the complete model system then transported to the test section. An identical electrical hookup would be provided in the preparation area and the test section.

2. The extremely high load cases may require a one-piece sting, necessitating transportation of the complete model and 12 foot long sting from the preparation area to the test section.

3. Model changes in the tunnel can be accomplished by extending an access tube into the tunnel. The tunnel will include a localized model heating system. With the exception of the working area in the test section, the tunnel will be maintained at a low temperature. The high cost of cooling emphasizes the fact that model changes will be extremely expensive, and that configuration development in a cryogenic tunnel will be minimized. In addition, the more complex models discussed previously must be reasonably trouble-free, again emphasizing the need for a very thorough pre-test checkout of all model systems, possibly under cryogenic conditions (static).

4. It has been shown that model handling and configuration changes in a cryogenic tunnel can be very expensive because of the cooling-down and warming-up sequences. It may therefore be worthwile to consider two similar models. The advance use of N/C in model manufacturing may result in a relatively cheap second model allowing one to be tested while the other is modified.[48]

C. Balance Check Calibrations in a High Pressure Tunnel

1. Afger some discussion, it was agreed that 50% of full load should be applied. For example in high load cases in the NTF this is 5000# to 8000#. Special provisions will be needed in the test section to apply such a load.

2. Support system deflections need to be verified, and this can be done during the check loading. Consideration must be given to the change of modulus at low temperatures, and this must be accounted for.

D. Auxiliary Flow in a High Pressure Tunnel

1. The need to test complex models with engine simulation has been discussed. Provision of the necessary auxiliary flow systems is mandatory, to provide inlet and exhaust simulation in those test programs where transonic flight conditions are a major design consideration. A large portion of the B-1 windtunnel test program, for example, was comprised of inlet and afterbody testing. The necessity of providing transonic exhaust simulation for launch and shuttle vehicles is questioned, however, since in the context of other major design considerations, the transient transonic operating regine would not seem likely to be of critical importance.

   At transonic test conditions, exhaust suction will be desirable for increased inlet mass flow. This requirement should be considered from both the model and facility points of view.

E. Model Filler Materials in a Cryogenic Tunnel

1. Regarding the development of model filler materials, it is believed that such effort is of major importance to the successful utilization of the cryogenic transonic tunnel. Filler material must possess the characteristics of tenacious adhesion and quick removal. Application should be simple and cure time short.

CONCLUSIONS AND RECOMMENDATIONS

It is highly recommended that coordination in the field of model systems be continued. This need is emphasized by the fact that subsonic high Re tunnels are now operational in Europe, that the National Transonic Facility in the USA will be operational in 1981, and that a decision on LEHRT is forthcoming.

Research in this field on both sides of the Atlantic appears to be increasing, and a sound plan for coordinating that effort is mandatory. The work-sharing plan described in this report forms the basis for a unified effort. Certainly it should be our goal to eliminate as many problem areas as possible prior to operation of the new high Re tunnels.

* * * * * * *

WORK SHARING PLAN
ACTION ITEMS

Listed below are the tasks identified as specific problem areas, that are worthy of special effort. Where applicable the country or agency planning, or performing research in a specific area is identified. The column "no planned action" indicates an area that deserves attention. Areas of high priority are those that are considered to be important enough to merit research on as broad a base as possible. In these cases consideration should also be given to establishing a meeting of the working specialists, where individual research efforts can be compared, reviewed and discussed, for the benefit of all participants and their respective agencies and countries.

Task	USA Canada	Europe	No Planned Action	Area of High Priority
Session 1				
A4			•	
B4			•	
D4			•	
D7	•			
D8	•			
F4	•			
F5			•	
G3	•			
Session 2				
A1	•	*		•
A2	•	*		
Session 3				
A1	•	•		•
B1	*	*		
C1	*			
C6	*			
C7			•	
Session 4				
A1, A8	•	•		•
A5			•	
B1	*			
B2	*			
B4			•	
B6			•	
B7	•	*		•
C1	*			•
F2			•	
G1			•	
H1			•	
K1		•		
Session 5				
A1	*			
B	*			
E			•	•

* Work planned for the future dependant upon available funds.

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APPENDIX 6

DESIGN OF TRANSONIC WORKING SECTIONS

by

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1. INTRODUCTION

Under the auspices of the AGARD Fluid Dynamics Panel Subcommittee on Windtunnel Testing Techniques and in harmony with the recommendations of the second report of the MiniLaWs Working Group, AGARD-AR-83, meetings of various experts on the Design of Transonic Working Sections were held. Representatives of European and USA and Canadian agencies held meetings almost simultaneously on February 23, and 24, 1976, at NLR, Amsterdam, and Calspan Corporation, Buffalo, New York, respectively. A subsequent meeting was held on May 25 and 26, 1976, at ONERA, Chatillon, between the co-conveners of the two groups and the representative from the United Kingdom who was unable to attend the previous meeting.

The meetings were concerned with discussion on the research activities affecting the design of transonic working sections which have been accomplished since the information in AR-83 was compiled. In addition, the group began to formulate a generalized plan to attack the wall interference problem. A summary of the discussion with conclusions and recommendations is presented herein.

2. CURRENT RESEARCH

Although these activities are summarized in Section 5, it is worth including here some of the comments put forward at the two meetings of the working group.

Theoretical Studies

The relationship between the Davis and Moore and the Chen and Mears modeling of the slotted wall boundary condition has been clarified by Barnwell.[1] The Chen and Mears rod-doublet representation in effect forms a dividing streamline around the slat, Figure 1, reducing the effective slot width. Barnwell interprets the boundary condition in terms of the radius of curvature of the dividing streamline at the slot location, R , and the wall porosity, δ/a . As the slat end radius approaches zero (an inherent assumption in the Davis and Moore analysis) the two formulations are shown approaching one another, Figure 2.

Significant advances in the development of numerical relaxation schemes as applied to transonic wall interference effects have been reported by Murman,[2] et al., with the TSFOIL computer code, by personnel[3,4] of NASA Langley, and by Kacprzynski[5] of NRC, Canada. However, these procedures must be used with caution. Steinle, NASA Ames, noted some numerical instabilities had been encountered with the TSFOIL code at large values of wall porosity. Kemp, NASA Langley, reported that because of mass addition the use of non-conservative finite difference formulations, such as the original Newman-Klunker[4] coding, yields large errors in the flow field away from the model. An example of the difference in the solution given by the conservative and non-conservative formulations is shown in Figure 3. It is ironic that the non-conservative formulations tend to predict experimental shock locations on windtunnel models better than the conservative formulations.

Steinle, NASA Ames, is formulating a theoretical representation of a test section and plenum to investigate the effects of plenum configuration on blockage buoyancy. Source panels and vortex sheets are being used to represent the plenum and test section walls, respectively. The formulation will employ the homogeneous slotted and porous boundary conditions. Data from three airfoil tests in the 2 x 2 ft tunnel will be used in support of the work.

Following the lead of the several researchers examining the adaptive wall concept in two dimensions, Lo and Kraft, AEDC, have begun preliminary work on the 3-D adaptive wall problem. Their examination of the 2-D theoretical formulation has led to an analytical proof-of-convergence of the inner and outer flow field solutions. In addition, an analytical expression for the optimum relaxation factor has been found for one modeling technique.

Experimental Studies

Flow Quality

The primary function of the transonic test section ventilated wall is the generation of uniform flow. Accordingly, this characteristic is one of the first properties which should be evaluated for any new wall geometry. Reference 6

presents data on five wall geometries consisting of a thin-wall variable-porosity specimen, an axially-distributed-porosity perforated wall based on the results of Reference 7, a rod-wall similar to that developed by the National Bureau of Standards, a low-noise modification to the six-percent inclined-hole perforated wall,[8] and a slotted wall based on the design being considered for the National Transonic Facility (High Re). Measurements from a centreline static pressure pipe and total pressure were used to calculate the longitudinal Mach number distribution. All of the walls yielded basically the same subsonic axial flow uniformity with an evident improvement being noted with walls which had long smooth transition regions from the solid to the ventilated portions. The rod and slot walls were clearly superior at $M = 1.1$, all walls were basically the same quality at $M = 1.2$ and the perforated walls were better at $M = 1.3$. Model blockage effects on the tunnel calibration were better at $M = 1.3$. Model blockage effects on the tunnel calibration were examined for all walls by placing a 2% blockage cone-cylinder around the static pipe. It was shown that a significant change in the approach Mach number occurred at values of less than 3% porosity with the perforated walls and all slotted wall porosities. However, at porosities above 3% with the perforated configurations, no effects of model blockage were observed. The flow downstream of the model returned to the calibrated free-stream conditions for all wall configurations and porosities.

Of the three modes of flow unsteadiness, turbulence, noise and temperature spottiness, noise appears to be the most important in transonic testing.[9] Noise levels of only .35% and .6% Cp rms were found to affect the transition location about 4% of the chord on a supercritical airfoil but did not change the trailing edge or shock induced separation.

A study of the relation between flow quality and the time required to achieve a given level of accuracy for various measurements is presented in Reference 10. Quantitative requirements for turbulence level as well as pressure fluctuations have been developed.

Current work at NLR on noise and its effects on turbulent boundary layers development and separation leads to the conclusion of a decreasing influence of noise with increasing Reynolds number. It was suggested that a slightly blunted cone (nose radium 0.5 mm) be used to study the effects of turbulence on transition and to detect transition by cone surface fluctuating pressure measurements rather than by probing the boundary layer. The NLR cone and associated equipment may be offered for use in other facilities.

The RAE is studying the possibility of designing working sections for transonic windtunnels which will have small dynamic interference effects on the relatively large models used for flutter and buffet tests. The dynamic interference effects of acoustic resonances and excessive noise within the working section are being considered. Preliminary tests have been made in a small windtunnel using circular cylinders operating in the subcritical range to provide a source of pressure fluctuations. The propagation of the pressure fluctuations away from the cylinder has been studied at Mach numbers up to 0.5. The preliminary results both with respect to the elimination of acoustic resonances and the reduction of noise in the empty tunnel at Mach numbers up to 1.0 are encouraging.

Wall Boundary Conditions

Correction for wall interference effects are not routinely applied to data taken in ventilated transonic tunnels in spite of the large volume of related theory and the need for such corrections. One of the reasons corrections are not applied is a lack of knowledge of the proper boundary condition to describe the tunnel walls in calculations of the interference.

Redeker, at DFVLR, is using different size models of the NACA 0012 airfoil to study the slotted wall boundary condition with slotted walls of 2 to 10% porosity. The airfoil data are being compared with theoretical computations, which include the effect of the boundary layer displacement, to assess the wall interference.

Smith reported good agreement is obtained on the boundary condition determined by the drag balance method and lift curve slope following the NLR internal note AC 75.14. However, with differences of pressure between the plenum and the test section, it seems better to measure the perturbation components at some distance from the wall.

A new test technique has been developed[11] to determine the pressure/flow angle relationship for any ventilated wall within the constraints imposed by the small perturbation assumptions. To avoid the necessity of obtaining high accuracy flow angularity measurements, an indirect approach was selected wherein sufficient other diagnostic measurements would allow calculations of the compatible flow angles. The relaxation method of Muram and Cole[12] provides the calculation tool to solve the two-dimensional nonlinear transonic small perturbation equation, subject to boundary conditions derived from static pressure measurements as indicated in Figure 4. Preliminary results show the true wall boundary conditions to be nonlinear, spatially variant and model dependent.

The measured pressures and inferred flow angles have been used to compute the boundary layer growth over (and the mass flux through) a perforated wall, subject to a variety of model disturbances. The ratio of displacement thickness to hole diameter for typical experimental configurations may vary in order of magnitude, e.g., 2.5 to .25, with axial position. It is anticipated that analysis of the results, still in progress, will produce a semi-empirical correlation of the mass flux with the pressure drop across the wall and local boundary properties. Given the correlation, it would then be possible to calculate the wall boundary conditions for any model-imposed pressure distribution.

The results from tests of models of the 11 ft and 2 ft transonic windtunnel walls at NASA Ames using an average mass flow technique indicate a classical wall porosity parameter, R, is not a function of Mach number. A technical note describing the experiment and presenting the results is forthcoming.

Experimental data from a small perforated Ludwieg tube tunnel[13] were obtained with varying wall boundary layer displacement thickness. Static pressures on a cone-cylinder model were compared with data from a conventional tunnel for $0.95 \leq M_\infty \leq 1.15$ to infer that a factor of two variation in the wall displacement thickness ($0.12 \leq \delta$/hole diameter ≤ 0.28 at one tunnel station) results in an equivalent wall porosity change of about one percent.

Experimental Wall Interference

Experiments with an idealized wing-tail pressure model utilizing a variety of wall configurations have recently been completed. Typical pressure distributions on the wing upper surface are shown in Figure 5. The major conclusion of the study is that the transonic wall interference is practically independent of wall geometry provided the ventilation is homogenous. The finding is applicable to all of the slotted, fixed porosity, and variable porosity, walls tested. There apparently is no wall geometry of uniform porosity which will yield interference-free data at transonic speeds with models of reasonable size (1% blockage). Axial variation of wall geometry as predicted by Lo[14] can influence the wing shock position as shown in Figure 6. The variations explored were obtained by using perforated wall sections with differing hole inclination angles. The most notable shift in the wing shock position for the condition shown resulted from the use of a solid-wall section directly above the model. The optimum wall configuration, however, was found to be a function of Mach number and model attitude, hence probably also model configuration.

A study of the drag on short conical bodies[15] has yielded additional insight into the wall interference problem near Mach 1. It was found that the percentage of drag reduction with increasing solid blockage is nearly identical to that reported by Couch and Brooks[16] for long slender bodies of revolution.

Reference 17 presents a study of the experimental effects of varying wall porosity (1.3 to 10%) on the pressures and forces of a 10% thick 2-dimensional supercritical airfoil at transonic speeds and high Reynolds number. The results show the effect of porosity is significantly larger at transonic than at subsonic speeds, the effect of Reynolds number (7 to 30 million) on the wall interference is small, and that conventional subcritical AGARD wall corrections[18] are, in general, inadequate at both subsonic and transonic speeds. The NAE wall matching method[19] yielded reasonable Mach number corrections but the incidence corrections did not appear to be uniformly applicable. However, Mach number and angle of attack adjustments did appear to neutralize the porous wall interference effects even though the proper values could not be determined a priori. It was also noted that significant differences in the far field pressure distribution both above and below the airfoil can occur at transonic speeds without seriously affecting the measured airfoil characteristics.

ONERA Model Correlation Tests

Analysis is not yet complete of data from tests of the ONERA M3, M5, and C5 models in the NASA Ames 11-TWT and the AEDC 16T and 4T wherein extreme care was taken to have exact similitude in the models and test conditions. Some conclusions, however, are evident. The effects of Reynolds number at $M = 0.84$ with free and fixed transition are summarized in Figure 7 wherein the variation of angle of attack, axial force and pitching moments at constant values of normal force are presented versus Reynolds number. For a given model, the data with fixed transition are more sensitive to Re than with free transition. Further, the variation of the data tunnel-to-tunnel is greater with fixed transition. Close examination of Figure 7 will reveal that the data from 4T are essentially the same whether transition is free or fixed, whereas there is a significant variation in the larger tunnels. Even though state-of-the-art manufacturing tolerances were used in model manufacture, the pitching moment data indicate the M3 and M5 models evidently have a slight difference in tail incidence which created difficulties in attempting model-to-model comparison. Analysis of the M5 wing pressure data from 16T and 4T with fixed transition shows that the data difference are manifested by spanwise dependent differences in shock-boundary layer interaction and trailing edge separation. An example is presented in Figure 8 wherein the leeward surface wing pressure at $x/C = 0.1$ is presented as a function of angle of attack along with inferred separation patterns at three spanwise stations. It is the investigator's reluctant opinion that because the ONERA model data are very sensitive to the state of the local boundary layer and the apparent lack of similarity between models, they may not be suitable for evaluation of wall interference until tunnel flow qualities are significantly improved. The results do very dramatically show, however, the necessity of evaluating the susceptibility of model data to tunnel flow quality before attempting to lump all data discrepancies measured between two tunnel tests into a single category.

3. PLANNED RESEARCH

New or Modified Facilities

A new blowdown transonic windtunnel will be in operation during 1976 at DFVLR. The test section will be 0.6 by 0.34 meters. Maximum stagnation pressure will be 5 atm. The work planned for the tunnel concerns noise level measurements and the efficiency of the settling chamber.

The design of a new blowdown facility is underway at FFA which will have convertible slotted/perforated walls in a test section 1.5 by 1.5 meters. Maximum stagnation pressure will be 7 atm. The tunnel will be used for 3-dimensional and half model tests. No decision, however, has yet been taken on whether or not to build the tunnel.

AFFDL has under construction a 9 x 9-inch transonic blowdown facility with a 48-inch long rod wall test section having the capability of non-ventilated contouring or variable distributed porosity. Research efforts are planned to investigate call corrections, wave attentuation, acoustic and support interference problems. Shakedown tests are projected to begin in December 1976.

The NASA Ames 2 x 2 ft tunnel is being modified for 2-dimensional wall interference and wave attenuation studies of various slotted wall configurations. Work is planned with NACA 64A010 and 0012 airfoil sections. Follow-on work with 3-dimensional configurations is anticipated.

Flow Quality

Studies to understand the mechanisms of and methods to suppress wall generated noise for several wall configurations are continuing at both AEDC and NASA Langley.

Measurements are planned for the AEDC transonic facilities and the FDL transonic windtunnel to determine the spatial distribution (x, y, z) of Mach number and flow angularity in the test region, the mechanisms producing excessive gradients if they exist, and means for reducing same. Spatial uniformities of 0.001 in Mach number and 0.05 deg in angle of attack are considered reasonable goals. Initial tests in the FDL program will concentrate on various types of flow angle sensors.

Tests are planned at ONERA to determine the effect of suppression of the wall generated noise by gauze on flutter and buffet onset with different planform models. In addition, tests are being considered to determine if gauze is effective in eliminating perforated wall edge tones over the full range of normal velocities through the holes.

Wall Studies

Calspan plans to resume 2-dimensional iteration for the moderately subcritical flow case but with subcritical flow at the tunnel wall ($M = 0.725$), $\alpha = 2$ and 4 deg) using present accuracy criteria. Both multipole and vortex distribution methods[20] will be used to define the unconfined conditions. All of the available data will be analyzed to determine practical convergence criteria with further experiments if necessary. Experiments will also be conducted for the subcritical wall case to determine the influence of variable distributed porosity within the segmented plena. The work will then be extended to the case with supercritical flow at the tunnel boundary ($M_\infty = 0.85$ and 0.90 with 2- and 4-deg incidence), testing new accuracy criteria with the transonic small disturbance methods. The influence of variable distributed porosity will also be assessed for the supercritical wall case.

For the adjustable solid wall configuration, Goodyer plans to extend work with the NACA 0012-64 airfoil to higher Mach numbers, higher lift coefficients, into stall, and perhaps to higher blockages. The next step at ONERA, after a short attempt on a 3-D case, is to apply the concept to the NACA 0012 airfoil at the Mach numbers and angles of attack of the Calspan experiment.

ONERA also plans to test an "optimum" 2-D test section configuration to reduce interference developed from theoretical parametric studies[21]. The new 2-D configuration has a solid lower wall and a low porosity upper wall.

AEDC plans to continue development of the adaptive wall concept with emphasis on the 3-dimensional problem. Initial efforts will be directed toward software development and numerical studies with follow-on experiments in future years. Continued evaluation of new ventilated wall configurations is also planned.

NASA Langley plans the continued refinement of 3-D rectangular transonic codes with both slotted and perforated wall boundary conditions. Experimental efforts will include investigations of 2-D contoured walls and rod walls in the 16 x 19-inch tunnel, 3-D slotted and rod walls in the pilot model of the National Transonic Facility and slot flow diagnostics in the 8-ft transonic pressure tunnel.

AFFDL plans to continue obtaining baseline pressure and flow angularity data for use in development of adaptive wall control logic. The next series of tests in the trisonic facility will use axisymmetric models at zero and small angles of attack. In addition, FDL is supporting research by Dr John Lee, Ohio State University, in the use of isolated plena (top isolated from the bottom) as a means of wall interference reduction.

The slot flow model developed at the FFA[32] is being extended to the 3-D and non-steady cases. This was reported at ICAS 1976 in Ottawa. Subsequently the study will be oriented toward the use of pre-computed, model-adapted slot geometries, monitoring the wall interference by measured wall pressures.

4. CONCLUSIONS AND RECOMMENDATIONS

While progress is being made, much of the current research work is in the 'crawl before walking" stage with specialized problems receiving, in some instances, considerable attention. Nevertheless, it should be kept in mind that the overall goal of the effort is to arrive at a best working section configuration for testing models of reasonable size over a Mach number range from about 0.6 to 1.2 and angles of attack which include large regions of separated flow. Each researcher should be cognizant of that goal in the design and accomplishment of his particular facet of the problem.

Wall Interference

It was unanimously agreed that the most pressing problem in transonic working section design is to understand the mechanisms and develop means for relieving wall interference with supercritical flow over lifting models. In many instances, particularly with large regions of supercritical flow, theoretical data corrections are impossible because there is no equivalent unconfined flow field to that experienced by the model in the windtunnel. For example, if for an aircraft, interference effects cause the wing terminal shock system to be significantly displaced, corrections to Mach number, incidence, curvature, etc. will not result in the empennage being in a comparable unconfined flow field. It is felt the adaptive wall concept holds the greatest promise for solving that problem. Given the present state of knowledge, however both numerical and physical experiments in both 2-D and 3-D are justified and should be pursued with vigor.

For the 2-dimensional case concept demonstration should continue into the highly supercritical flow range. When possibility has been shown, work should proceed to improve the speed and minimize the program size for the flow field calculations, establish reasonable convergence criteria, and make systematic variations of the tunnel geometrical parameters including test section length to optimize the various configuration variables.

The first step for the 3-dimensional case is obviously the selection or development, if necessary, of computational techniques for calculation of the flow field from the interior measurements. Interior measured quantities, their number, locations, and required accuracies need investigation. Test section shapes which are not square should not be discarded a priori. Obviously, concept demonstration experiments must be performed following essentially the same sequence used in the 2-D work, but with the added complication of how to treat the sidewalls. It is recommended that exploratory experiments be designed to determine if solid or passive slotted sidewalls can be used in the 3-D case to allow the use of shadowgraph, schlieren, or laser techniques with the adaptive wall. The possibility of correcting for side wall effects if they are not fully adaptive or of using contoured sidewalls should be explored. Once feasibility has been demonstrated, systematic variable-optimization experiments and computational refinements should be performed.

It should not be construed, however, that adaptive wall configuration studies are the only fruitful areas for investigation. Studies of the basic wall interference mechanisms, disturbance generation and propagation, wall boundary layer effects, wave/wall interactions, etc. which will provide a better phenomenological understanding of flows throughout the transonic Mach number range are still needed.

Standardization

In order that wall development work have a common basis, standard models, test techniques and instrumentation must be adopted whenever possible. Toward that end, the following comments and recommendations are made, with the research objectives of Section 4.1 in mind:

(1) It is unfortunate that transonic airfoil sections are inherently sensitive to the effects of both Reynolds number and tunnel flow quality. Therefore, it is recommended that both 2-D and 3-D initial wall development work be accomplished with the NACA 0012 airfoil which seems relatively insensitive to both effects. Once a basic solution has been found, more representative airfoils should be tested.

(2) While excellent base line force and pressure data from the Calspan 8-ft tunnel are available in tabular form for a 6-inch chord, 0012 section, manufacturing tolerances are never sufficient to duplicate model contours to the precision required for model to model comparisons. Therefore, it is recommended that interference free data be obtained on each model to be used in wall development work in the transonic range. In addition, both 2-D and 3-D model surfaces should be sufficiently hardened and corrosion resistant to prevent erosion during testing, handling and storage.

(3) It is recommended that a standard 3-D model which is easily mathematically modeled be designed and adopted from the following criteria:

   (a) A small as practicable cylindrical centrebody with an ogive nose.

   (b) A NACA 0012 airfoil wing at zero incidence with respect to the centerbody and zero taper, sweep angles of both zero and 25 to 35 deg would be useful to investigate the effect of lift on axial interference gradients.

   (c) An instrumented empennage – at least the horizontal tail with a constant chord 0012 airfoil and sweep similar to the wing.

   (d) A standardized sting configuration. The model instrumentation should include a direct measurement of Mach number and angle of attack.

(4) It is recommended that where possible both pressure and force measurements be obtained simultaneously. Such a practice will clearly be impossible in the smaller facilities. In those cases, it is recommended that initial development work be done with pressure models because so much more information can be gained therefrom. Pressure instrumentation for the 3-D model must include at least three spanwise stations on the wing and two on the horizontal tail in addition to orifices along the centerbody. The model or sting must be instrumented to allow incidence corrections caused by load deflections in the pitch plane. Provisions for the measurement of unsteady pressures would also be useful.

(5) Test Reynolds must, of course, be determined by the capabilities of each facility. Nevertheless, once detail design of the "standard" 3-D model has been established, tests with both free and fixed transition should be conducted over a range of Re in more than one large facility to assess the configuration's susceptibility to Re and flow quality effects. State-of-the-art methods of fixing transition should be used.

(6) In addition to model data, pressure measurements in the inviscid portion of the flow field near the tunnel boundaries and wall boundary layer data would be useful for validating interference flow field models and assessing the "correctability" of the data. Flow visualization of the transonic flow fields is recommended whether the investigation concerns adaptive or conventional walls to aid in interpretation of the data.

(7) The responsibility for a detailed design study leading to a standard model for 3-D wall interference evaluation should be undertaken by a single agency rather than an interagency committee.

Wall Boundary Conditions

Both theoretical and experimental work to define the true boundary conditions of the variety of walls in present use should continue. Such information will allow a more accurate evaluation of the interference in and the limitations of present facility, provide guidance toward selection of an adaptive wall configuration, and possibly lead to a decrease in the number of iterations required for adaptive tunnels. The condition of the local boundary layer as influenced by model imposed pressure gradients throughout the flow field must be considered in such research.

Correlation Studies

It is recommended that more 2-D and 3-D tunnel-to-tunnel correlation studies, such as the ONERA model program, be undertaken in order to gain more insight into, as yet, not fully identified transonic testing problems. Models with extensive pressure instrumentation would be preferable over force models. In every case it is necessary to provide along with the experimental results as many details of the flow conditions in the windtunnel as possible. Correlation testing with both fixed and free transition should be done with the tunnels Reynolds numbers adjusted to be consistent with the findings of the cone correlation study.[22] Extensive pressure measurments in the inviscid portion of the flow field, particularly in the vicinity of the tunnel boundary, would be most useful in providing data for comparison with and development of theory.

Flow Uniformity

In most tunnels it is assumed that if the centerline Mach number distribution is satisfactory and model integrated pitch plane flow angularities are low and flow throughout the test region is also satisfactory. While such an assumption is probably justified in low subsonic flow, a growing amount of generally unpublished data indicate it is not necessarily true for transonic speeds. Therefore, it is recommended that spatial Mach number and flow angularity measurements be made in the test region of existing tunnels to identify problem areas so that corrective action can be taken if necessary.

Noise and Turbulence

While significant advances in wall-generated noise-suppression devices have been made, work should continue to assess the effect of such devices on other wall properties. Experimental data providing criteria to establish acceptable tunnel noise levels for various types of measurements and sampling times is needed. The influence of noise and turbulence on boundary layer development is still a fruitful area for research.

Related Problems

Although not a topic for consideration by the Working Group, we urge that:

(1) Work should continue to define the effects of both the amplitude and frequency of acoustic noise and turbulence on transonic aerodynamic phenomena in order to help establish criteria for acceptable flow environments in transonic tunnels.

(2) More definitive work be done on the effects of boundary layer transition devices (grit, glass beads, etc.) on the development of turbulent boundary layers with both favorable and adverse pressure gradients.

(3) Methods be found to compensate for the effects of model supports. Such data would be most helpful in sorting out phenomena which could be erroneously construed as wall interference effects.

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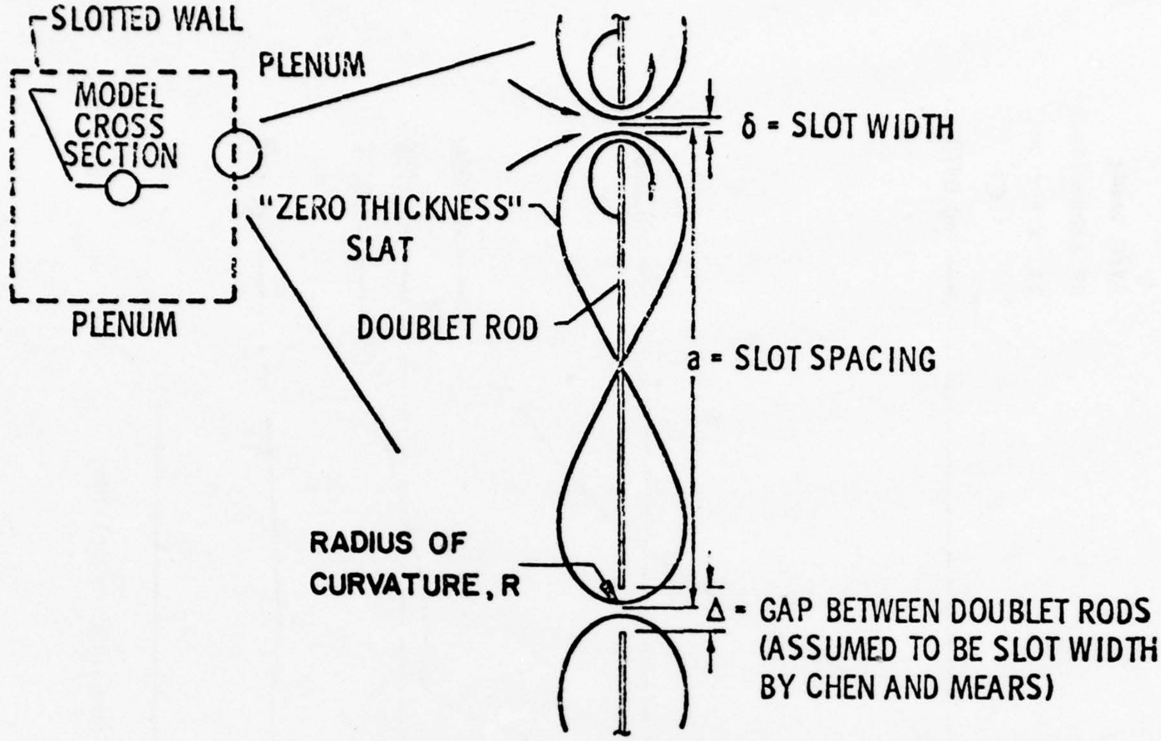

Fig.1 Chen and Mears model of slotted wall

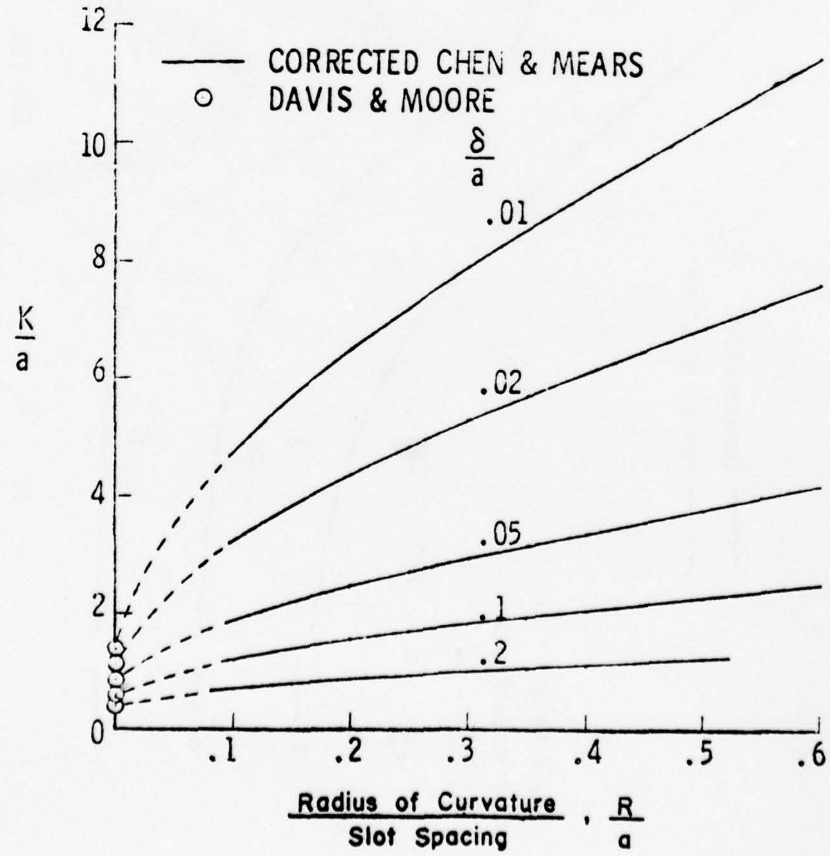

Fig.2 Correlation of slot parameter K and slot radius of curvature R

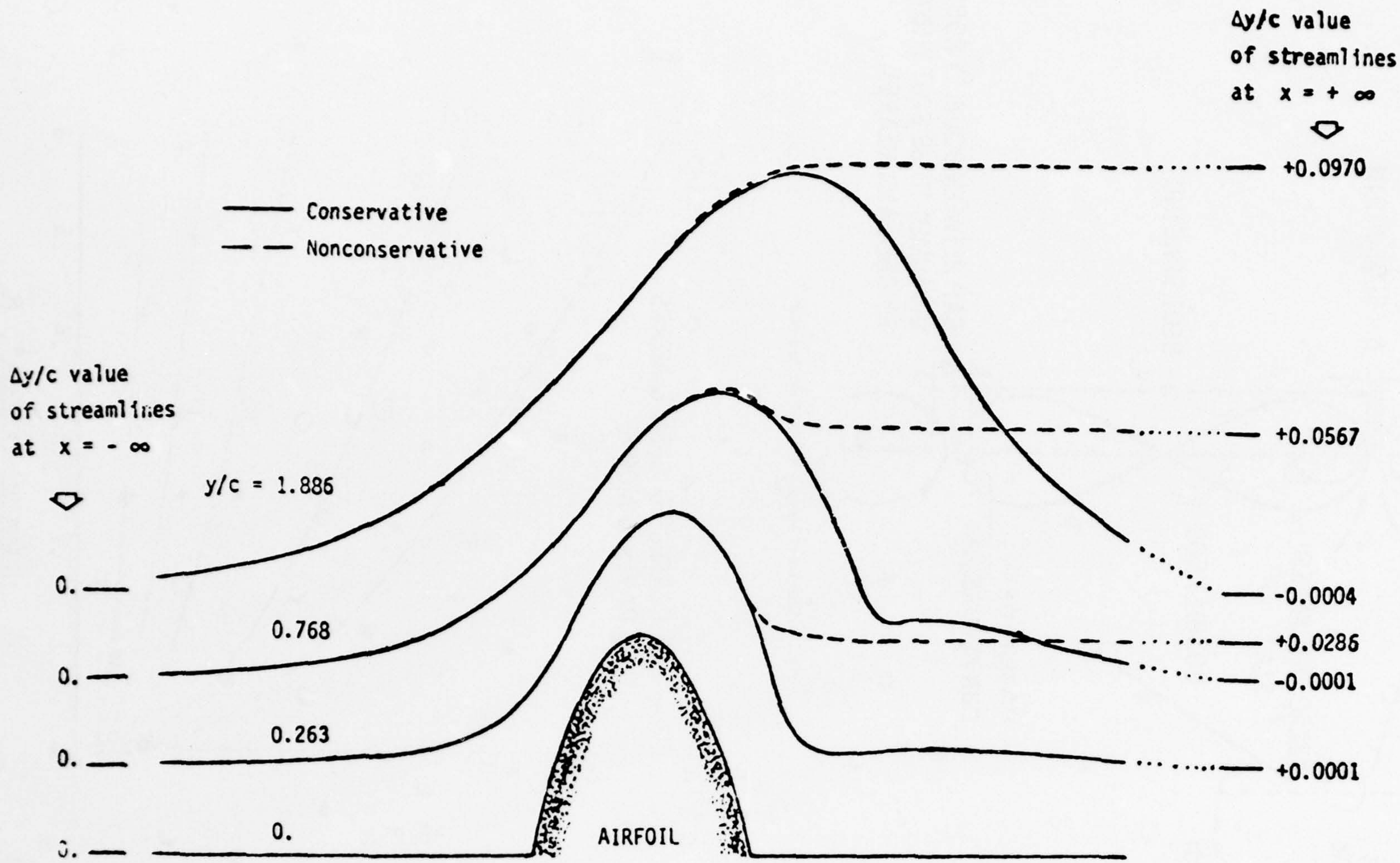

Fig.3 Effect of finite difference formulation on calculated free-air streamline deflections in transonic flow, $M_\infty = 0.95$

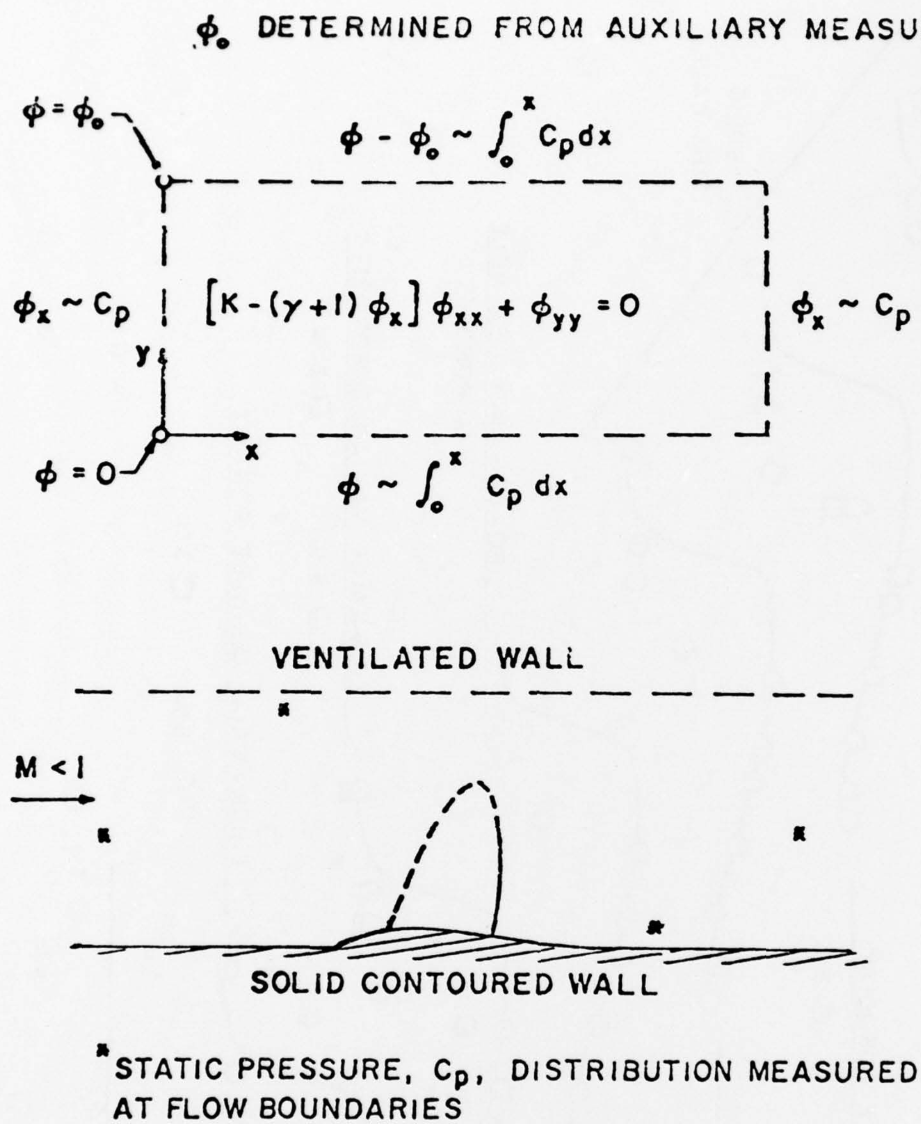

Fig.4 Technique for determining the true ventilated wall boundary condition

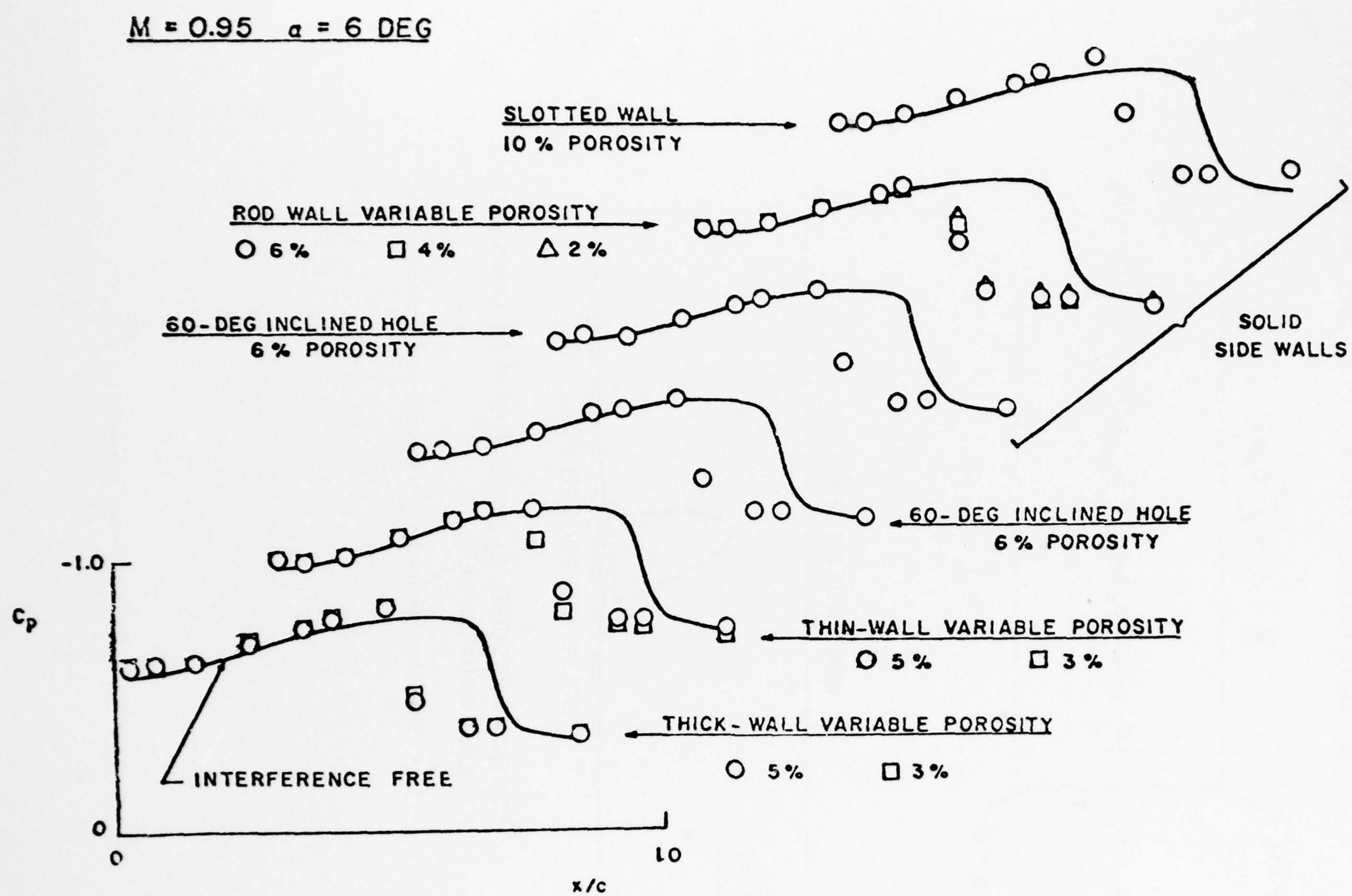

Fig.5 Upper surface wing pressure distributions resulting from several wall geometries of uniform spatial dimensions

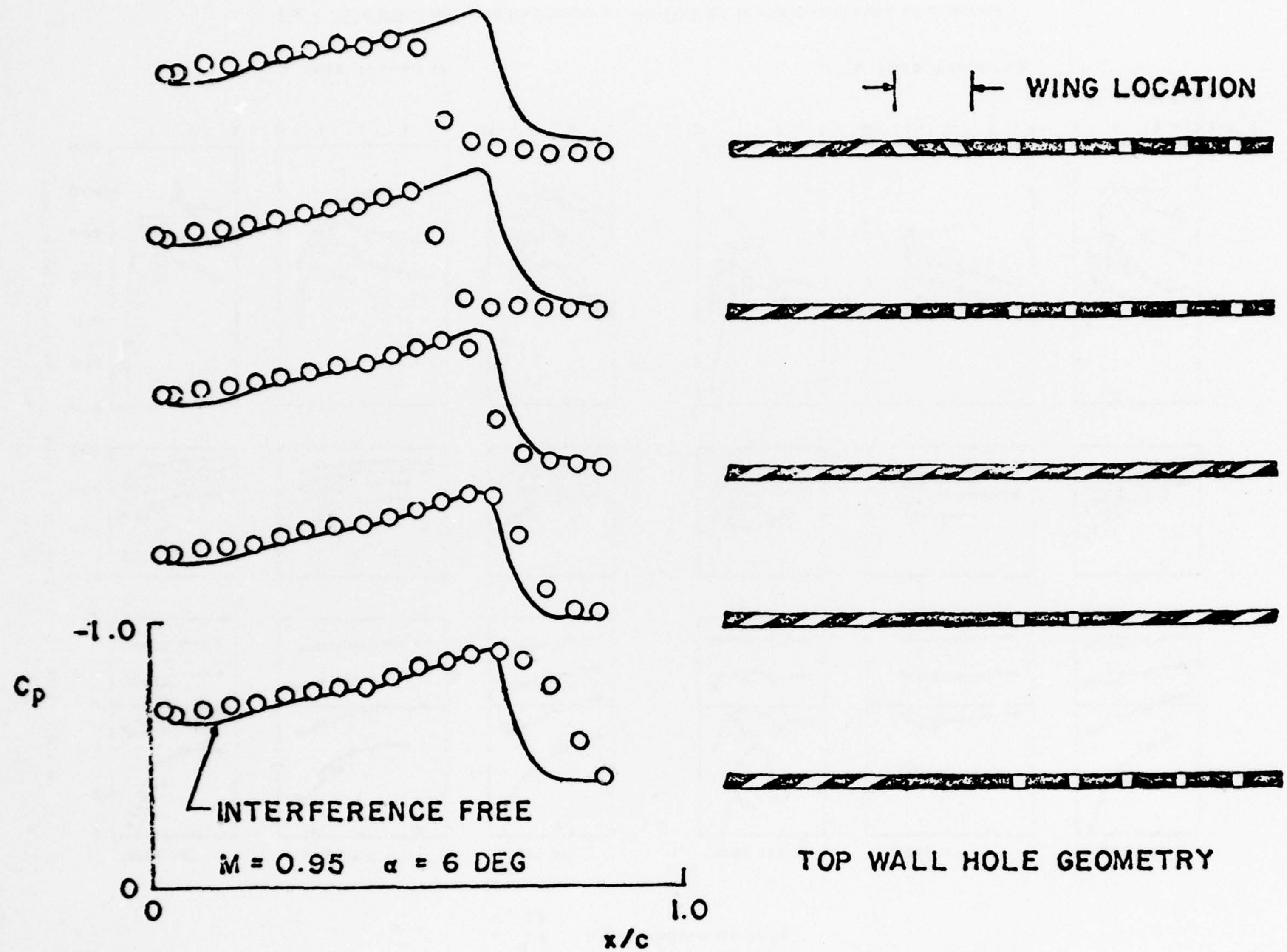

Fig.6 Effect of axial variation of wall geometry on the upper surface wing pressure distribution

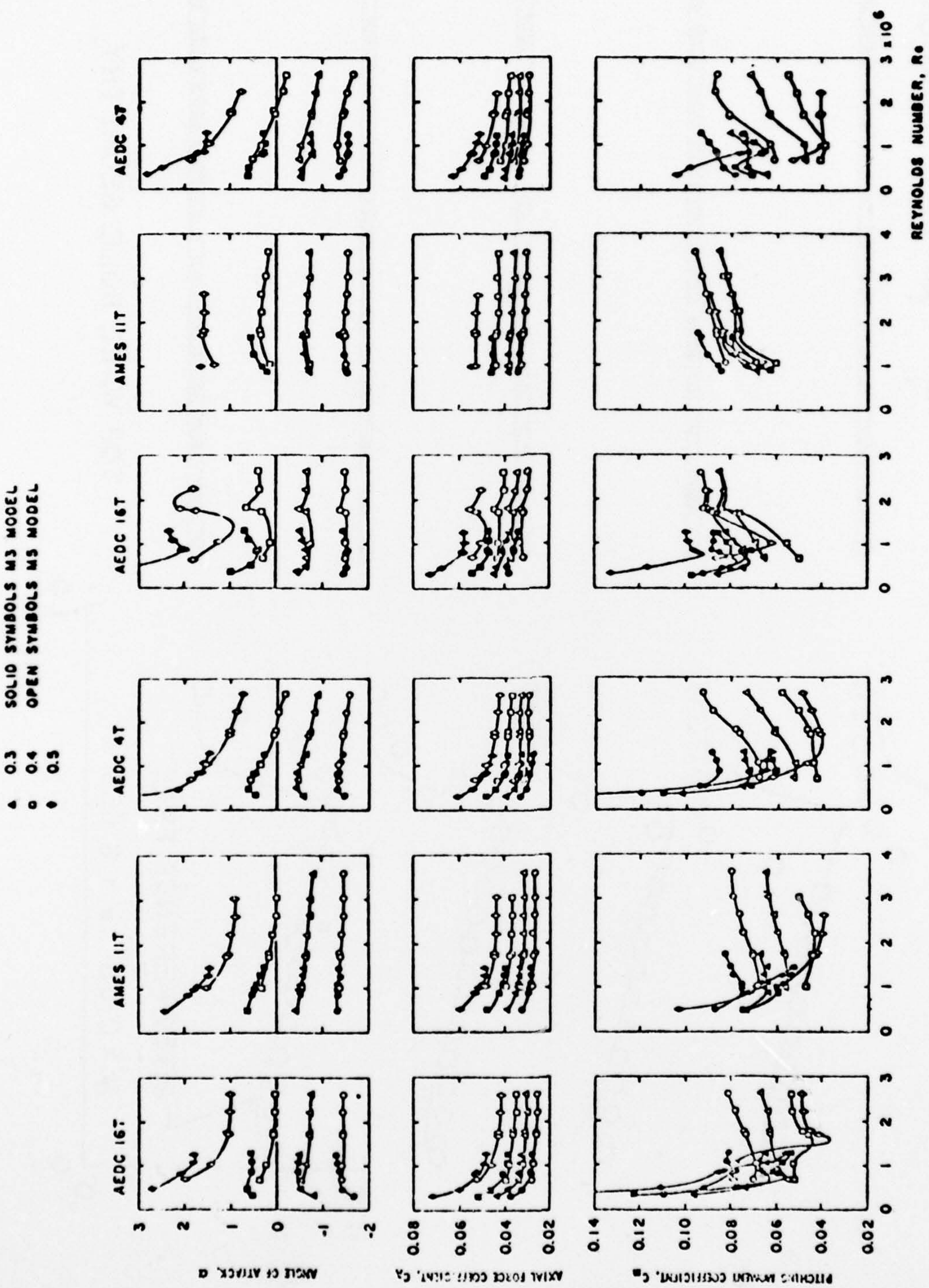

Fig.7 The effect of Reynolds number on the ONERA model data in three windtunnels

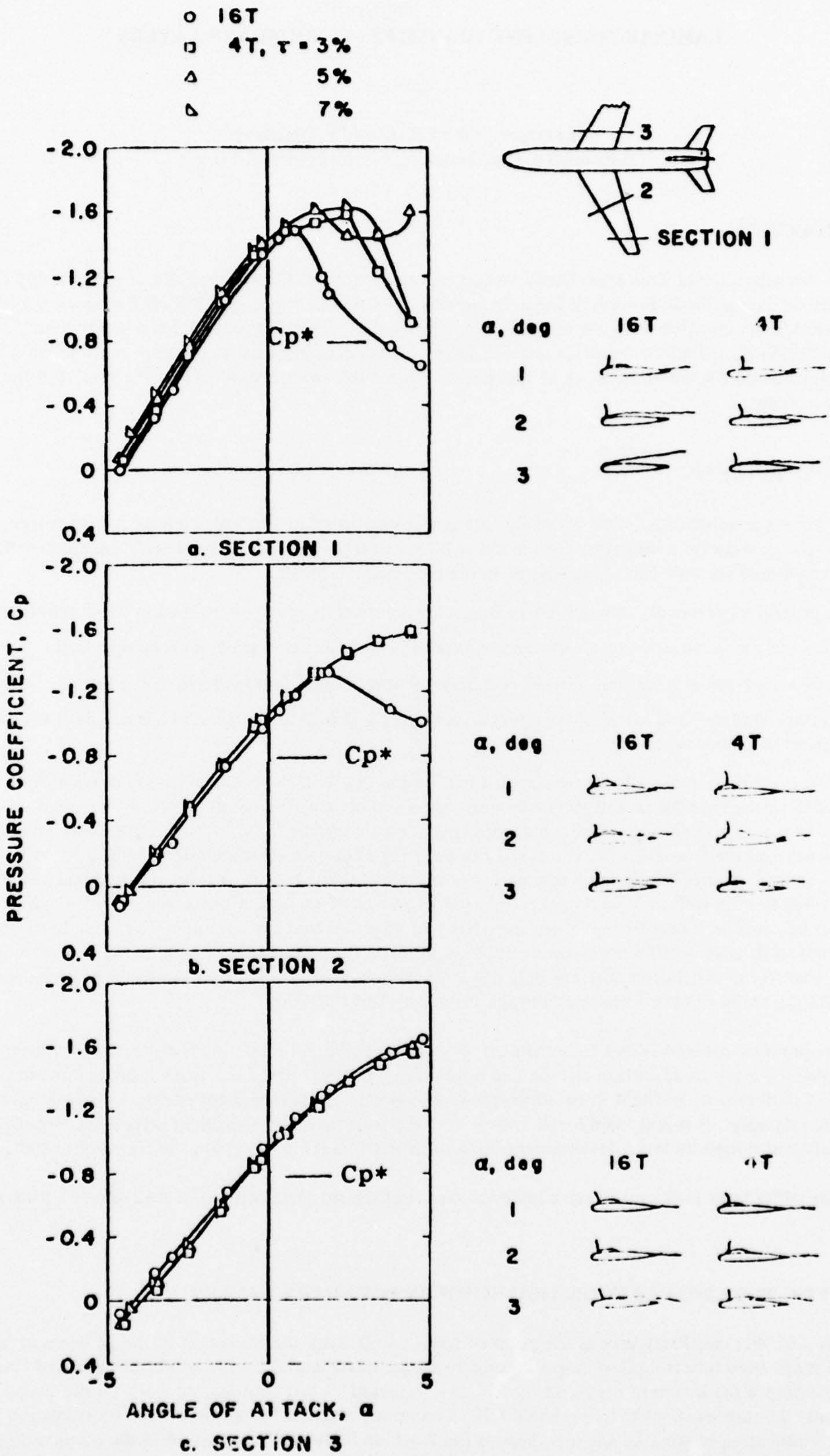

Fig.8 Pressure at x/C = .01 and separation patterns on the M-5 wing in 16T and 4T, $M_\infty = 0.7$

APPENDIX 7

LAMINAR-TURBULENT TRANSITION IN BOUNDARY LAYERS

by

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1. INTRODUCTION

The US Boundary Layer Transition Study Group represented by its Chairman, E.Reshotko, and the Eurovisc Working Party on Transition in Boundary Layers represented by its Chairman, E.H.Hirschel serve as consulting groups for the AGARD Subcommittee on Windtunnel Testing Techniques. The US group met the last time on July 13, 1976, and the Eurovisc Working Party on June 21 and 22, 1976. The chairmen of the two groups met on April 26 and 27, 1976, in Cleveland and discussed the work of the groups. In the following a summary of the results of the meetings and discussions is given.

2. GENERAL RESULTS

Developments in aeronautics show more and more that viscous effects will play a large role in future aerodynamics. These effects which were for a long time considered as higher order effects in ordinary aerodynamics (with few but important exceptions) are now becoming key phenomena in many respects:

(a) In general, airplanes fly at much larger Reynolds numbers than can be reached in windtunnels today.

(b) With the supercritical wing, a very complicated viscous interaction problem is being posed.

(c) Fuel conservation in aeronautics will certainly become a significant problem in the future.

(d) Military airplanes and missiles will operate increasingly at high angles of attack (post stall) where large separated flow areas exist.

In all these instances the viscous effects are coupled predominantly with the boundary-layer development. Even if full scale models at the right Mach and Reynolds numbers could be tested in windtunnels, design goals could be reached only to a certain degree since the windtunnel poses an environment for turbulent boundary layer development that is different from the free-flight situation, and the handling of certain effects could only be achieved by very expensive parametric studies because not enough is known today about turbulent three-dimensional boundary layers and their development as well as possible laminarization by means of suction. Considering only the many possible transition mechanisms on a swept wing[1,2] one even has to conjecture that the turbulent boundary layer on the wing or at least parts of it, will be different in structure, depending on the mechanisms which led to its creation. All these points lead to the conclusion that not only must significant efforts be given to the study of turbulent boundary layers but also the study of their initiation through instability and transition.

Another point of concern is that today almost all work on turbulent modeling is done with windtunnel data. This is true also for work on transition criteria and prediction methods. Since this work is being done in order to be able to predict performance in flight in the atmosphere knowledge of the free-flight environment will have to be pursued more intensely. It is reasonably well known how the windtunnel environment affects the transition process as well as turbulent boundary layer development but almost nothing is known about the free-flight situation.

Certainly other large problems in aerodynamics exist, but the problems discussed here must be considered as very basic.

3. EUROVISC WORKING PARTY ON TRANSITION IN BOUNDARY LAYERS

The Eurovisc Working Party uses as the point of focus for its work the transition on swept wings at transonic speeds. The scope therefore is limited in speed range to incompressible and compressible subsonic and transonic flow. At the last meeting many different problems have been considered[3,4], but coordinated work on the swept wing problem is only developing slowly. In September 1975 recommendations for research work on transition were published.[3] Publication of some of the members of the Working Party will be reviewed in the forthcoming Eurovisc Report 1976 (Ref. 6).

Both theoretical and experimental work on transition in two-dimensional flows especially at ONERA/DERAT make the following ideas apparent:

(a) For the moment, the transport equations for turbulent flow are a transition prediction tool of high potential. The mean flow properties in the transition region can be calculated with reasonable accuracy while taking into account free-stream turbulence, pressure gradients, and general boundary conditions. The amount of empiricism to be introduced is relatively small, especially at the basic level.

(b) On the other hand, experimental observations gave new insights which clearly indicate that the methods mentioned above will have to be superseded by new methods with much different modeling.

For three-dimensional flow the situation is different since on swept wings especially, the mechanisms leading to transition are entirely different from those in two-dimensional flow. The difficulties already encountered for two-dimensional flows suggest that the approaches to be followed be at least systematic as those followed in the two-dimensional case.

(a) Acquisition of empirical criteria.

(b) Work on basic problems such as stability which will lead to a better understanding of the flows as such.

(c) Finally perhaps also modeling of transport equations, arriving subsequently to higher levels of understanding and prediction capability.

Considering the new techniques available and in development for experimental and theoretical work, it should be possible to make progress although too much optimism should be avoided. From the side of the design aerodynamicists, demands will arise in any case for a better understanding of the phenomena and for better prediction methods.

4. US BOUNDARY-LAYER TRANSITION STUDY GROUP

The US Boundary-Layer Transition Study Group was founded in late 1970 (as the NASA Transition Study Group) to develop and implement a program that would do something constructive toward resolving the many observed anomalies in boundary layer transition data and that might provide some basis for future estimation of transition Reynolds numbers. The group formulated specific experimental programs emphasizing careful and redundant measurements, documentation of the disturbance environment and eliminating wherever possible, facility induced transition. It recommended continued study of stability characteristics as well as theoretical studies of the coupling of various types of disturbances to boundary layers.

Reports of the initial results of the programs of the Transition Study Group appeared in 1974 (Refs 7,8). These papers pointed out the need for careful measurement and careful interpretation of results. Even experienced investigators were found not to be infallible.[9] The effects of tunnel sound on transition signature and on transition behavior generally were clarified for supersonic Mach numbers between 3 and 8. The mechanisms by which tunnel sound excited disturbances in a supersonic boundary layer was identified but not yet in all of its details.[10,11] The previously observed unit Reynolds number anomaly in ballistic ranges[12] was extensively rechecked[13] but not resolved. The results of investigations aimed at developing a high Reynolds number "quiet" supersonic windtunnel were presented.[14,15] Such tunnels would operate without the dominating influence of the sound radiated by turbulent boundary layers on the tunnel walls. Last but not least it was shown how disturbance environment could be incorporated into a transition prediction procedure[10] providing the first step toward development of a rational method for transition prediction. An account of the foundations of the program of the US Boundary Layer Transition Study Group appears in a review article by its chairman.[16]

Presently the Transition Study Group is undertaking two additional programs both of which are in the advanced stages of planning. The first is a flight-test of a cone at $M \approx 2$. This is a cone that has already been extensively tested in windtunnels all over the world. The proposed tests should provide some clues on the relationship between flight environment and tunnel environment. The second program is on the fundamental processes underlying the effects of small bluntness on transition. Work continues on the attempts to develop rational procedures for transition prediction.

For purposes of record, the US Boundary Layer Transition Study Group has since the fall of 1974 operated under the auspices of the Arnold Engineering Development Center of the US Air Force.

5. SUGGESTED AREAS OF INVESTIGATION

Based on results to date, the following areas of further work are suggested:

Basic Mechanisms

(1) Linear Stability – Expand the catalog of normal modes results through exact numerical solution of the appropriate sets of disturbance equations. Obtain experimental corroboration as necessary or desirable. Most available results are for zero pressure gradient cases including heating and cooling but with constant wall temperature. Factors that require consideration are: three-dimensional effects such as those introduced

by angle of attack and sweep, bluntness, suction and blowing, pressure gradient and longitudinal temperature variations.

(2) Receptivity – The understanding of the coupling of radiated sound to a boundary layer is fairly well understood. More must be done however both by theoretical modeling and by careful experimentation to acquire understanding of how free-stream turbulence, entropy disturbances, roughness, model vibrations, etc. introduce disturbances into a developing laminar boundary layer. In all cases including that of tunnel sounds, it is important to understand how a forcing disturbance of a given frequency and prescribed phase velocity excites a free disturbance of the same or related frequency but of different phase velocity.

Nonlinear Processes – Further understanding is needed of all aspects of behavior subsequent to the growth of infinitesimal disturbances. Some of the important features of non-linear growth are the changes in frequency and orientation spectra of disturbances through distortion of the mean flow, generation of harmonics, mode-mode coupling, secondary instabilities, etc. leading perhaps to turbulent spot formation and eventual transition. From the viewpoint of transition prediction, the important contribution from non-linear studies might be in the development of amplitude criteria to be used in the various transition prediction procedures now being developed.

Transition Prediction

All transition prediction theories, whether based on linear stability considerations or on turbulent model equations, should have ways of incorporating input disturbance information. It is also desirable that the amplitude criterion developed for and compatible with a given method have some physical and/or correlation basis. One might also contemplate at this point a turbulence model procedure as a follow-on to a linear stability calculation.

Transition Testing

In experimental studies of transition, there should be continued improvement in experimental technique displaying sensitivity to potential difficulties and care in overcoming them. It is recommended that such work be conducted within the following guidelines as formulated by the US Boundary Layer Transition Study Group.

(1) Any effects specifically and only associated with test facility characteristics must be identified and if possible avoided. This points to emphasizing studies in ballistic ranges and "quiet" tunnels.

(2) Attention must be given to disturbances introduced by model surfaces, model material, and internal structure. Experimental studies should include documentation of these various factors.

(3) Details of coupling of disturbances of various kinds to the boundary layer must be understood theoretically and experimentally, so that the sensitivity of the transition process to the flight environment might be determined.

(4) Whenever possible, tests should involve more than one facility. Tests should have ranges of overlapping parameters, and whenever possible, experiments should have redundancy in transition measurements.

6. CONCLUSIONS

The importance of viscous effects in aerodynamics is fully appreciated nowadays. While numerical methods exist for prediction of the properties of laminar and turbulent boundary layers, the methods currently being used for transition prediction have severe limitations. Much work, both theoretical and experimental, is needed to provide the basis of a rational prediction procedure for boundary layer transition. However deficient our knowledge is for two-dimensional flows, it is even more so for three-dimensional flows such as encountered for example on swept wings.

Assuming that a better capability of transition prediction is desirable for future aerodynamic design, it is necessary to define a research program utilizing the best methods and techniques available today; and which makes use of the possibilities of coordinated work in different countries on both sides of the Atlantic Ocean as outlined in the text and references of this Conveners report.

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14. Abstract

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Part I describes the work of the TES Subcommittee and gives the rationale for the effort. Part II reviews the research that is underway and gives comments and recommendations derived from that review. These comments and recommendations are the principal contributions of the TES Subcommittee members. Part III summarizes the main conclusions and recommendations. Part IV lists titles, investigators' names and locations for the research and studies that are reviewed herein.

Seventy-nine leading research workers from nine countries participated in the work of the TES Subcommittee and made valuable contributions.

Need for advances and the possibilities for achieving further technology gains are developed in this report and sixteen technology gains requiring further research are specified in Part III.

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